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PROJECT HAZEL

AERODYNAMICS

REPORT NO. ZA-282

OCTOBER 1958

CONVAIR

A DIVISION OF GENERAL DYNAMICS CORPORATION

SAN DIEGO, CALIF.

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REPORT ZA-282
DATE 10/27/58
MODEL Hazel

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PROJECT HAZEL

Aerodynamics

Report ZA-282
Date 10/27/58

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FOREWORD

This report is presented as one of a set describing the Project "Hazel" study performed by the Convair San Diego Division of the General Dynamics Corporation. The entire set of reports, listed below, represents Convair's fulfillment of the publications obligation specified in Contract NOas-58-812 (SS-100) and Amendment #1, issued 14 August 1958 by the Bureau of Aeronautics.

ZP-252 Summary (Brochure of Charts with Text)

ZP-253 Aircraft Design

ZA-282 Aerodynamics

ZJ-026 Propulsion, Structure Heating, and Pressurization

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SYMBOLS

A_3 = engine reference area - ft^2

A_6 = engine exit area - ft^2

A_c = capture area - ft^2

C_D = drag coefficient

C_{D_i} = induced drag coefficient

C_{D_O} = zero lift drag coefficient

$\Delta C_{D_{warp}}$ = increment in C_D due to wing warp

C_L = lift coefficient

C_{L_δ} = lift coefficient due to control deflection - 1/deg.

C_M = control effectiveness - 1/deg.

C_M = pitching moment coefficient

C_{M_O} = pitching moment coefficient at zero lift

C_N = normal force coefficient

C_n = yawing moment coefficient

C_{n_β} = yawing moment coefficient slope - 1/deg.

c.g. = center of gravity

c.p. = center of pressure

$\bar{c}$ or MAC = mean aerodynamic chord - ft.

g = load factor - acceleration of gravity

H.M. = control hinge moment - lb-ft.

ℓ = reference length = root chord - ft.

L/D = lift to drag ratio

M_A = moment of area - ft^3

M = Mach number

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p	=	pressure - lbs/ft^2
q	=	dynamic pressure - lbs/ft^2
RN	=	Reynolds number
S_{Ref}	=	wing area - ft^2
S_v	=	fin area - ft^2
S_c	=	control surface area - ft^2
SF-1	=	liquid hydrogen
$S_{L.E.}$	=	leading edge cross sectional area - ft^2
S_B	=	wing base area - ft^2
SFC	=	specific fuel consumption - lb/hr/lb
S_T	=	area of control tab - ft^2
t/c	=	thickness ratio
V	=	velocity - ft/sec.
V_1	=	average velocity during cruise on first 1/3 of fuel - ft/sec
V_2	=	average velocity during cruise on last 2/3 of fuel - ft/sec
Δx	=	distance along X axis from c.p. to c.g. - ft.
$X_{a.c.}$	=	distance from leading edge of MAC to aerodynamic center - ft.
$X_{c.p.}$	=	longitudinal distance from leading edge of MAC to the point through which the resultant aerodynamic force acts - ft.
$X_{c.g.}$	=	longitudinal distance from leading edge of MAC to c.g. - ft.
W_0	=	initial weight - lbs.
W_1	=	weight with 1/3 of fuel consumed - lbs.
W_2	=	weight with all fuel consumed - lbs.
W_f	=	weight of propellant - lbs.
α	=	angle of attack of root chord - degrees
α_T	=	trim angle of attack - degrees

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δ = control surface deflection - degrees

δ_T = control surface deflection required for trim - degrees

β = sideslip angle - degrees

γ = ratio of specific heats

$\Lambda_{L.E.}$ = sweepback angle of wing leading edge - degrees

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SUMMARY

This memorandum presents the aerodynamic characteristic, performance and methods of analysis used in the study of several manned reconnaissance vehicles.

Studies were made to size a number of configurations to yield ranges of 3,200 or 4,000 nautical miles carrying payloads of 300 to 1,300 lbs. A basic configuration designated MC-10 was selected and a more detailed analysis of lift, drag, stability, control and maneuverability was made.

The basic characteristics of MC-10 are included in Table I.

TABLE I

Payload	lbs.	800
Range	N.Mi.	3,200
Cruise	Mach number	3.0
Cruise altitude	ft	125,000 to 137,800
Weight and start of cruise	lb	13,800
Fuel (Pentaborane)	lb	6,330
Lift/drag at start of cruise	L/D	4.17
Reference (wing) area	ft^2	1,985
Engine - 1 Marquardt ramjet		

The MC-10 meets or exceeds the performance requirements stated on page 1 of this report.

A three-view of MC-10 is shown in Figure 1.

Additional configurations are shown in Figures 2 through 6.

The basic characteristics of a number of ramjet cruise configurations are shown in Table II.

Velocities required of boost-glide and boost-rocket cruise-glide type vehicles were found unacceptable by the customer. As a result these configurations are included for record purposes only and are not discussed in detail.

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INTRODUCTION

The purpose of the work summarized in this memo was to conduct a preliminary study of a manned reconnaissance vehicle, either glide or cruise, to meet the following requirements:

Reconnaissance altitude	150,000 to 200,000 ft (100,000 ft. minimum)
Cruise speed (if cruise type)	Mach No. 2 to 3.
Glide speed (if glide type)	as low as possible
Range	3,200 n.mi. (2,500 min.)
Reconnaissance range at optimum altitude	1,500 n.mi. (1,000 min.)
Payload	800 lb. (400 minimum)
Crew	1
Seabased	
Wing loading, W/S	<10 lb/ft^2

This report covers the aerodynamic studies that were made in obtaining a vehicle that meets the stated requirements. Boost-glide, boost-rocket cruise-glide and boost-ramjet cruise configurations were evaluated.

The minimum range considered was 3,200 n.mi. and the minimum altitude, 125,000 ft. at start of cruise while carrying a payload of 800 lbs. at Mach number = 3.0.

The body of this report discusses the aerodynamic characteristics, performance and sizing of vehicles with alternate numbers and type of power plant, alternate fuel types, range and payload.

A detailed discussion of the methods and equations used in determining aerodynamic characteristics and performance is included in Appendix A.

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DISCUSSION

1. Types of Vehicles Studied

Three general types of manned reconnaissance vehicles were considered initially. These were: boost-glide, boost-rocket, cruise-glide and boost ramjet cruise.

The first two were studied briefly and appeared quite feasible from the standpoint of design problems, size, materials and range and altitude potential. The basic characteristics of the rocket powered configurations are summarized in Table III. A typical three-view is shown in Figure 7.

These configurations were rejected by the customer, apparently because the average velocities were too high for the reconnaissance mission and severely limited maneuverability.

The operational characteristics of the boost-ramjet cruise type vehicles were found acceptable, and considerable effort was expended in studying configurations with alternate ramjets, fuel types, number of engines, fuels, payload, range and cruise altitude.

2. Aerodynamic Design Limitations Imposed by Operational Requirements

The aerodynamic performance of the ramjet powered configurations was compromised by the requirements of a fairly flat, clean undersurface with engines, pilot and payload located on top of the wing and within the limits of the leading and trailing edges.

Additional penalties were incurred by a requirement (Convair imposed) for a manual control system and static stability subsonically. The additional trim drag incurred by designing for static stability subsonically resulted in an 11.5% increase in gross weight and a decrease in cruise altitude of 3,000 ft. This was considered a reasonable penalty to pay for the improvement in simplicity and reliability.

3. Trajectories

The flight of the ramjet powered vehicles consists of three phases: the boost phase, the cruise phase and the glide phase. The boost phase consists of boosting the vehicle to the design cruise altitude at a Mach number of 3.0. The range during boost is conservatively assumed as 20 n.mi. During the ramjet cruise the vehicle is designed to fly a Breguet range path and therefore the altitude increases as fuel is burned. The

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cruise range is either 3,000 n. miles or 3,800 n. miles, depending on the vehicle. When the fuel is exhausted the vehicle begins a maximum L/D glide. The range in glide is approximately 180 n. miles. Figure 8 presents an altitude range plot for 3,200 n. miles and 4,000 n. miles configurations. The time from launch is plotted at several points along the trajectories.

Booster studies were assigned to Boeing Airplane Company by the Navy project coordinator under separate contract. For this reason no detailed studies of the boost phase were made by Convair.

Limited hand calculations indicated that a vehicle of the Hazel type, launched from a B-36 at 45,000 ft., would require a booster mass ratio of about 1.88 to reach Mach 3.0 at 125,000 ft. For the MC-10 configuration this led to a launch weight of 30,525 lbs assuming metal parts of the booster equal to 17% of the booster propellant weight.

4. Ramjet Powerplant Characteristics

The characteristics of the ramjet powerplants proposed by Marquardt and Pratt and Whitney are discussed separately. Only circular, pod type engines were considered.

The engines proposed by Marquardt are constructed of a non-metallic, light weight material in any size required. Optimum burner length is stated as 16 feet with large penalties in thrust and specific fuel consumption (SFC) for shorter lengths at altitudes above 100,000 ft. Inlets are circular and of the internal-external compression type resulting in optimum range characteristics. Performance is quoted with Pentaborane or Hydrogen (SF-1) fuels. Figures 9 thru 11 are reproductions of data provided by Marquardt and show thrust coefficient, C_T and SFC at varying altitudes and Mach numbers. Engine weight is found from Figure 12 and external drag from Figure 13.

The engines proposed by Pratt and Whitney are of conventional metallic structure, and the maximum engine size is limited to approximately 8 ft outside diameter. The size limit is imposed, since P & W is unwilling to promise an engine that cannot be tested in existing facilities. The burner length of P & W engines is much shorter than that of Marquardt designs, with no explanation of this difference at present. Performance is quoted with Pentaborane or SF-1 fuels. Fuel economy burning Pentaborane is poorer, while that for SF-1 is better, than quoted for the comparable Marquardt engine. Figures 14 and 15 present plots of the Pratt and Whitney engine data used in this study.

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5. Aerodynamics

This section consists of a brief discussion of the aerodynamic characteristics of the ramjet cruise vehicles. A detailed discussion of the methods and equations used in determining the aerodynamic characteristics is given in Appendix A.

The aerodynamic characteristics have been examined in detail at Mach 3.0 but only briefly at subsonic speeds.

It is noted here that, due to time limitations, the lift, pitching moment and trim characteristics have been evaluated only for a basic single engine vehicle and a basic twin engine vehicle.

Lift

The lift is made up of the lift of the wing and lift due to the internal flow in the ramjet. It has been assumed that the external lift of the body is zero due to the manner in which it is located in the flow field of the wing.

The wing has been warped to give minimum drag due to lift with a 10% wing static margin. The wing was optimized on the IBM 704 computer program of Reference 1.

Pitching Moment

The pitching moment characteristics have been evaluated at Mach number 3.0 with power on and at subsonic speeds with power off in order to find the center of pressure shift. At Mach number 3.0 the pitching moment consists of the pitching moment due to the body and wing normal force and a C_{M_0} term due to the zero-lift drag and engine thrust. The pitching moment at subsonic speeds is made up of the same terms as above except that the nose down moment from engine thrust is replaced by a nose up moment from engine drag. The center of pressure shift obtained above was found to be 13% of the MAC. In order to provide a stable subsonic vehicle with at least a 1% static margin, the forward c.g. was placed at .32 MAC. The trim drag at the start of the Mach 3.0 cruise was reduced to zero at the design C_L by placing the aft c.g. at .45 MAC. The fuel is programmed so that the c.g. remains at .45 MAC as the first third of the fuel is burned and then moves forward linearly to the .32 MAC point as the remaining fuel is used.

Controls

Tip elevon controls were selected for pitch and roll control after an examination of various types of surfaces. The tip controls have several advantages. They will not interfere with the engine inlet flow and they increase the possibility that the hinge moments can be reduced to be compatible with a manual control system. A tip control area of 1/9 of the wing area has been selected based on low speed characteristics. Directional

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control is provided by rudders on the vertical fins.

Drag

The drag is made up of three terms: the zero lift drag, the drag due to lift and the trim drag. The zero lift drag is made up of thickness drag, leading edge drag, base drag, friction drag, and a small increment due to wing warp. The friction drag has been evaluated at 135,000 feet. The trim drag is the increment in drag due to lift resulting from deflecting the control surfaces. A zero lift drag breakdown for the MC-10 configuration is presented in Table IV.

Directional Stability

The directional stability characteristics were analyzed and a fin size selected that would give a directionally stable vehicle over the complete flight regime.

6. Vehicle Sizing

The vehicles were sized to give the required cruise range of either 3,000 n. mi. or 3,800 n. mi. The cruise range was obtained using the Breguet range equation:

$$\text{Range} = .592 \left[(L/D)_1 \frac{V_1}{SFC_1} \log_e \frac{W_0}{W_1} + (L/D)_2 \frac{V_2}{SFC_2} \log_e \frac{W_1}{W_2} \right] \text{ N. Miles}$$

where

W_0 = gross weight, lbs.

W_1 = $W_0 - 1/3\ W_{fuel}$

W_2 = $W_0 - W_{fuel}$

V_1 = average velocity as the first third of fuel is used. - ft/sec.

V_2 = average velocity as the last two thirds of fuel is used - ft/sec.

$(L/D)_1$ = (L/D) aft c.g.

$(L/D)_2$ = average (L/D) as c.g. shifts to forward c.g.

SFC_1 = average specific fuel consumption as the first third of fuel is used - lb/hr/lb

SFC_2 = average specific fuel consumption as the last two thirds of fuel is used - lb/hr/lb

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The sizing of the vehicles to give the necessary cruise range involved an iteration procedure. The vehicles were assumed to start cruise at 125,000 ft., or higher if wing loading permitted. Actual starting altitude was determined by optimum L/D.

A gross weight was assumed and a lift coefficient that would give approximately $(L/D)_{max}$ was used to find the required wing area to start cruise at 125,000 ft. C_{D_0} was then evaluated assuming the ΔC_{D_0} contribution from the engine. The value of L/D at the start of cruise was evaluated and using this value the required engine thrust was obtained. The thrust required was marked up by 10% and the required capture area obtained using a thrust coefficient provided by the engine manufacturer. The engine drag was obtained and compared with the assumed value. If there was an appreciable difference the calculations were repeated using the new value.

The next step in the sizing procedure was a weight breakdown. The weight breakdown for all the ramjet powered configurations that were studied is shown in Table II. The ski weight is assumed 2% of the empty weight. The wing weight is obtained from Figure 16 as a function of gross weight and wing area assuming a non-rigid structure, pressure stabilized. The engine weight is obtained from Figure 12. The fuel system is assumed to be 10% of the fuel weights for Pentaborane fuel and 15% for SF-1. The difference between the assumed gross weight and the dry weight is the fuel weight.

Before the range equation could be solved it was necessary to obtain L/D at the forward c.g. position. When the c.g. is at its forward position at Mach 3.0 there is a trim drag penalty resulting in a 12% decrease in L/D.

The cruise altitudes corresponding to 1/3 fuel used and all fuel used were found, and these altitudes, together with the starting altitude, were used to determine the average velocities and average specific fuel consumptions for use in the range equation.

The range equation was then solved. If this did not give the required cruise range, the assumed gross weight was decreased or increased as required and the calculations repeated.

7. Configurations

This section discusses briefly each of the various ramjet configurations analyzed. All of these had a 60° delta wing with 2 inch diameter leading edge 3 inch diameter trailing edge. The engines are mounted on pylons above the wing. There are one or two vertical tails depending on the number of engines. The total vertical tail area is 14% of the wing area. Longitudinal control is obtained with tip controls having a total area equal to 1/9 of the wing area. The basic characteristics of all the analyzed configurations are presented in Table II.

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MC-10

The MC-10 was selected as the basic configuration and is shown in Figure 1. This selection was somewhat arbitrary though it does incorporate the feastures considered most desirable, as follows:

1. Single engine hidden above the wing.
2. Light weight and minimum volume through use of Pemtaborane fuel.
3. Flexible, non-metallic, pressurized structure.

The MC-10 is a 3,200 n. mile range vehicle powered by a single Marquardt ramjet mounted on a pylon over the wing root chord. The pylon contains the pilot, instrumentation and fuel. There are two vertical fins mounted outboard on the wing. The Pentaborane fuel is programmed so that the c.g. remains at .45 MAC as the first third of the fuel is consumed and then moves forward to the .32 MAC point when all the fuel is used. The basic structure is non-metallic, flexible and pressurized.

MC-11

The MC-11 is a fixed c.g. version of the MC-10. It is designed to have a constant c.g. at .45 MAC and hence no trim drag during cruise. It is therefore an unstable vehicle during the glide. This version was investigated in order to show the penalty involved in having a stable glide vehicle. This penalty amounts to a 1,400 lb. difference in gross weights.

MC-14

The MC-14 is a single engine Marquardt ramjet vehicle designed for 4,000 n. miles range. It weighs 27,000 lbs., has a 3,000 ft^2 wing and is basically a longer range version of the MC-10.

MC-15

The MC-15 is a 300 lb. payload version of the MC-10. The 500 lb. reduction in payload results in a 1,600 lb. reduction in vehicle gross weight to 12,200 lbs.

MC-17

The MC-17 is a 1,300 lb. payload version of the MC-10. This 500 lb. increase in payload results in a 1,615 lb. increase in vehicle gross weight to 15,415 lbs.

MC-19

The MC-19 shown in Figure 2 is a rigid non-metallic, pressure stabilized structure version of the MC-10. The wing and fins are made of rigid plastic rather than a flexible material. The rigid structure results in a gross weight increase of 4,500 lbs.

MC-20

The MC-20 shown in Figure 3 is a 3,200 n. mile range vehicle with two Marquardt ramjets mounted on pylons above the wing outboard of the root chord. The pilots compartment, instrumentation and fuel are contained in a body situated on top of the wing root chord. This body fairs into the single vertical tail. Gross weight is 13,150 lbs. The engines extend beyond the wing leading edge.

MC-22

The MC-22 is a 3,200 n. mile version of the MC-20 using SF-1 fuel instead of Pentaborane. The density of SF-1 is less by a factor of 8.4 than Pentaborane which means a much larger fuel tank is required and hence a higher C_{D_O}. The specific fuel consumption of SF-1 is much less than Pentaborane and the net effect is a reduction in gross weight to 8,600 lbs.

MC-24

The MC-24 is similar to the MC-20 except that it has shorter engines as shown in Figure 4.

The shortened engines have poorer performance which resulted in an increase in vehicle weight of 3,350 lbs. over the MC-20 to 16,500 lbs.

PC-20

The PC-20 is a 3,200 n. mile vehicle with two Pratt and Whitney ramjets using Pentaborane fuel. The Pratt and Whitney engines are metallic and have a greater weight per pound of thrust than the Marquardt engines. The specific fuel consumption is also greater. These two factors result in a vehicle 1,205 lbs heavier than the MC-20 for a gross weight of 14,350 lbs.

PC-22

The PC-22 shown in Figure 5 is an SF-1 fueled version of the PC-20. The Pratt and Whitney specific fuel consumption for SF-1 is less than the Marquardt value which helps to offset the heavier engine weight. The PC-22 is 1,100 lbs. heavier than the MC-22 for a gross weight of 9,700 lbs.

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PC-24

The PC-24 shown in Figure 6 is a 4000 n. mile version of the PC-22. It has two Pratt and Whitney ramjets and uses SF-1 fuel. It has an extremely large fuel tank which increases C_{D_0}. Gross weight is 13990 lbs.

8. Maneuverability

A brief analysis of the turn capabilities of these configurations has been made. An example has been calculated for the MC-10. It was assumed that in making a turn the vehicle dropped down in altitude so that it could complete the turn without increasing the angle of attack more than 2°. The two degree limit was established since thrust data are quoted for ± 2° angle of attack on the inlet, with thrust penalties at higher angles (-5% @ 3 deg.). The equilibrium altitude as a function of turn load factor is shown in Figure 17. A 180° turn was calculated starting at the point where 1/2 of the fuel has been used. The fuel used during the turn was subtracted and the remaining range calculated. The range for a flight with a 180° turn was assumed to be the range up to the start of the turn plus the range after completing the turn. Figure 17 shows the loss in range as a function of the load factor (g's) in the turn.

One-Engine-Out Performance of Two-Engine Configurations

The one-engine-out performance of the twin engines configurations was investigated. There is enough fin and rudder to trim out the yawing moment. However, the maximum thrust of one engine is less than the zero lift drag of the configuration, which means the mission can not be completed.

9. Parametric Studies

In the course of this project several parametric studies have been made and are discussed in this section.

Effect of Start-of-Cruise Altitude on Vehicle Size

Figure 18 presents a plot of vehicle gross weight vs start-of-cruise altitude for 3200 n. mile range. It is seen that the vehicle weight goes up quite rapidly with increase in starting altitude. A limiting start-of-cruise altitude of 140,000 feet is shown. This is due to the increase in specific fuel consumption with altitude, and to the decreasing mass ratios resulting from rapidly increasing wing weight.

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Effect of Changes in Fixed Weight (Payload) -

The effect of varying fixed weight (payload) on vehicle size for 3200 n. mile range is shown in Figure 19. It is seen that there is an upper limit of 2500 lbs. additional weight that can be carried 3200 n. miles. Any weight increases over this limit result in decreased range.

Range Parameter

The Brequet range equation shows that the range is directly proportional to the parameter $(L/D)\,\frac{V}{SFC}$. A plot of this parameter against altitude for MC-10 configuration at both forward and aft c.g. positions is presented in Figure 20. It is seen that the MC-10 is designed to fly very close to the maximum value of this parameter.

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APPENDIX A

This Appendix gives a detailed explanation of the methods of analysis used in computing the aerodynamic characteristics of the ramjet cruise vehicles. A detailed analysis was made on only the basic single engine vehicle, designated MC-10. To save time and man-hours the basic lift and pitching moment characteristics of the power plant used on this vehicle were used on all other configurations. The characteristics have been evaluated in detail at Mach 3.0 with only a brief analysis at subsonic speeds.

Assumptions were made where necessary, consistent with time and accuracy requirements of the study.

LIFT

The lift of the configuration is made up of the lift of the wing and the lift of the body or power plant. The following assumptions have been made with respect to the lift analysis:

1) The body had no effect on the lift of the wing.

2) The body lift consisted of the engine internal lift. The external lift of the body was assumed to be zero.

3) The thrust vector is assumed parallel with the flight path. This is conservative since the engine for the MC-10 is bent to give a component of lift equal to thrust times sin α.

Wing

The wing which was used on these configurations has been optimized for minimum drag using the method outlined in Reference 1. It was optimized at a Mach number of 3.0 to have minimum drag at a given lift coefficient, C_L, with a 10% static margin. The wing aerodynamic characteristics resulting from this program are presented in Figure 21.

These aerodynamic characteristics have been used directly for all twin engine configurations. In the single engine configurations, however, there is a flat ramp on the upper surface of the wing starting at the apex which goes back to approximately the chordwise location of the engine inlet. The purpose of this ramp is to keep the wing leading edge shock away from the inlet and to provide undisturbed flow at free stream conditions to the engine at the design angle of attack. The effects of this ramp have been approximated by applying shock expansion theory to a plane delta wing of the same planform with and without a ramp in the region of the design angle of attack. At these conditions it was found there was approximately a 7% loss in lift.

Body

The body as used in this report refers to the ramjet engine. In the single engine configuration the body is mounted on a pylon above the wing root chord. In the twin engine configurations the engines are mounted on pylons above the wing outboard of the root chord.

The body lift was calculated as the internal lift resulting from deflecting the inlet stream tube. The lift and normal force have been assumed equal. It has been assumed in this analysis that the external lift of the body is zero. The body is located in a complex flow field of the wing; a flow field which will vary considerably with angle of attack. A brief investigation has shown that in the region of the design angle of attack the forces on the external body shell are quite small. In future studies the external flow over the body should be investigated more thoroughly in order to verify these assumptions.

The body has a bend in it with the centerline of the front part inclined at -8.5° relative to the wing root chord. The aft part of the body is parallel to the root chord. This was done so that at the design angle of attack of 8.5° the inlet would be at a relative angle of attack of 0°.

The internal normal force coefficient was divided into two terms; a term at the lip and a term at the bend of the body. The term acting at the lip is given by

$$C_N = \frac{2.0}{57.3} \frac{A_C}{S_{ref}} (\alpha - 8.5)$$

where A_C = capture area.

The term acting at the bend is a constant given by

$$C_N = \frac{2.0}{57.3} \frac{A_C}{S_{ref}} 8.5$$

A plot of total lift coefficient against angle of attack at Mach 3.0 for single engine and twin engine configurations is presented in Figure 22. As previously mentioned, the difference in these two curves is due only to the affect of the ramp ahead of the single engine inlet.

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PITCHING MOMENT CHARACTERISTICS - MACH NO. = 3.0

The pitching moment consists of the pitching moment due to the normal forces of the wing and body and a C_{MO} term due to zero angle of attack drag and engine thrust. The vertical c.g. was assumed to be located 2.5 ft. above the upper surface of the wing.

Wing

The warped wing used on these configurations has a 10% static margin for the wing alone at the design lift coefficient. The pitching moment coefficient about the .5 MAC point is a constant for each design lift coefficient. The wing design lift coefficient was selected so that at the aft c.g. the complete configuration would have approximately a 1% static margin at a lift coefficient of .139 with no trim drag. The wing shape actually used for the MC-10 was optimized for a design lift coefficient of .09. This wing is not the optimum since a wing with a lower design static margin and a design lift coefficient closer to the configuration cruise lift coefficient would give slightly less drag due to lift. In follow-on studies the wing shape program should be re-run to take advantage of the lower static margin and higher design lift coefficient.

The induced drag of the wing was obtained from Figure 21. The wing pitching moment contribution was found using the vertical c.g. position given above.

Body

The pitching moment terms due to the body are given by

$$C_M = C_N \frac{\Delta x}{\ell}$$

where x = distance from c.g. to c.p. of component along x axis.
ℓ = reference length = 58.6 ft.

The c.p. of the internal lift term due to deflecting the flow at the inlet was assumed at the lip. The bend in the body was assumed to be located at the aft c.g. so that its moment arm was zero.

The C_{MO} term as used here will refer to the pitching moment coefficient due to zero angle of attack drag and engine thrust.

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The C_{M_0} due to drag was evaluated using the wing, fin and pylon C_{D_0} terms.

The C_{M_0} due to the engine was evaluated using the net thrust, (net jet thrust minus external engine drag).

PITCHING MOMENT CHARACTERISTICS - SUBSONIC

The pitching moment at subsonic speeds is made up of the same components as at Mach 3.0 except that there is no thrust term but rather an increased engine drag term due to the cold flow internal drag. A detailed analysis at subsonic speed has not yet been made.

Wing

The subsonic characteristics of the warped wing used on these configurations were not obtained. In order to approximate the subsonic characteristics the pitching moment characteristics of a plane delta wing were evaluated. Test data from Reference 2 gave an estimate of the C_{M_0} shift caused by warping a delta wing.

This C_{M_0} shift was then applied to the plane delta wing.

Body

The body normal force characteristics and centers of pressure were assumed to be the same as at Mach 3.0.

Two drag terms were used in evaluating the pitching moment coefficient due to zero angle of attack drag. These were the wing C_{D_0} and the engine off drag.

PITCHING MOMENT COEFFICIENT CURVES

The subsonic and Mach 3.0 pitching moment coefficients are plotted against lift coefficient in Figure 22. This figure shows that an aft c.g. location of .45 MAC is required in order to have approximately a 1% static margin at the design C_L of .139. This figure also shows that a forward c.g. of .32 MAC is required in order to have a stable subsonic configuration.

LONGITUDINAL CONTROL REQUIREMENTS

The possibility of using a manual control system for the Hazel was the governing criteria in defining the longitudinal control system. The control requirements were analyzed for three regions of flight; supersonic cruise, supersonic maneuvers, and subsonic flight and landing at zero fuel condition. The boost phase was not considered during this study as boost configurations were primarily defined in terms of performance. It is felt that the

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airplane balance during boost can be made compatible with the control system as defined from other phases of flight.

During an extended cruise with a manual control system, the airplane should be stable. Longitudinal control is required here to trim out the stabilizing moment at the cruise C_L and to handle c.g. shifts which occur as a result of fuel being consumed.

Some maneuvers are required during the supersonic flight region. Only moderate angle of attack changes can be tolerated since angle of attack is generally limited by its effect on the thrust and drag balance. To accomplish most maneuvers, a reduction in altitude will be necessary. Therefore, longitudinal control is required to handle only small changes in trim for supersonic maneuvers.

The amount of control required for subsonic flight following cruise will be governed by landing conditions. These generally involve high angles of attack and the most forward a.c. position.

SELECTION OF LONGITUDINAL CONTROL

Trailing edge flaps, tip controls and nose flaps were considered for pitch control.

After a cursory examination, the tip controls were selected as the most promising type. Sufficient control effectiveness could be obtained with moderate size control surfaces. They would not interfere with the inlet flow of the engine when deflected. Most important, there is a possibility that the hinge moments can be reduced so that they may be compatible with a manual control system. Structural characteristics of the wing were not defined sufficiently to evaluate the aeroelastic effects, therefore, no further consideration of this influence on control selection was made at this time.

TIP CONTROL CHARACTERISTICS

Experimental data on tip controls was used whenever possible to define the characteristics of this type of configuration.

Lift due to control surface deflection versus Mach number is shown in Figure 23. As indicated on this figure, the lift coefficient, based on control area is assumed independent of control size. This assumption is justified by examining the C_L vs control size at supersonic and subsonic speeds as illustrated in Figure 24. At supersonic speeds the variation is linear; at subsonic speeds slightly higher values of lift are obtained experimentally than those given by the linear approximation used.

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The control effectiveness, C_{M_δ}, is primarily a function of the moment of area of the control surface at supersonic speeds. Figure 25 illustrates this result at a Mach number of M = 1.61. Figure 25 also shows the variation of $x_{a.c.}/\bar{c}$ with M for the complete wing (i.e. $S_c/S_{Ref} = 1.0$). For the half delta tip control, the relation between moment of area of the control to moment of area of the wing is $M_A/M_{A_{wing}}$ and the control area to wing area is S_c/S_{Ref} These ratios (together with the linear variation of control force and moment with control area and moment of area, respectively) then define the relationship between control size, control deflection and center of pressure at supersonic speeds. Subsonic characteristics were obtained from experimental data and assumed to hold up to M = 0.8. A linear interpolation was used for $x_{c.p.}/\bar{c}$ values at $0.8 < M < 1.2$. These results are shown in Figure 26.

The normal force and center of pressure of the control itself are shown in Figure 27. The results are in terms of unit control area and the half delta tip control root chord. It should be noted that the center of pressure on the control due to α is forward of that due to δ at supersonic speeds. It appears feasible to balance the controls during cruise so that the hinge moments can be handled by the pilot. These results and comments apply to small deflections and angles of attack. With closely balanced controls, non-linearities in hinge moments are amplified. These non-linearities will have to be determined experimentally to accurately define control parameters.

Control Size

Since maneuvers during cruise will be accomplished by changing altitude and maintaining angle of attack nearly constant, there is no real requirement for control moment here other than handling the c.g. travel for trimmed flight. A more definite criteria can be used to size the longitudinal controls by considering the landing conditions. Figure 28 presents the minimum trim speed associated with control size and deflection for a c.g. at .32 $\bar{c}$. The most effective control at low speeds is the one which trims the airplane at a given velocity with the smallest deflection or conversely, trims to the lowest speed with a given deflection. Thus optimum control size for low speed flight is established from this figure as being 10% to 13% of the wing area. The ratio selected was

$$S_c/S_{Ref} = 1/9$$

It should be noted that if the actual c.g. location for low speed flight is further aft, the optimum control size can be reduced.

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The control deflection required for trim and the change in α per unit δ as a function of c.g. position at the cruise Mach number are presented in Figure 29 for the control size selected.

Control Balance

Although it is proposed to closely balance the controls at supersonic cruise (see section on "Tip Control Characteristics"), the large size of the surface and the large deflections required for trim at the forward c.g. location still produce large hinge moments. At cruise conditions with the surface balanced within 1% of the control root chord, hing moment per degree deflection is:

$$H.M./\delta = 85 \text{ ft. lbs./deg.}$$

The large control c.p. shift as Mach number goes subsonic also increases hinge moments. A trailing edge flap on the controls is proposed to handle the trim moments and change in hinge moments with Mach number. The flap would be operated two ways, as a trim tab and as an anti-balance tab.

The schematic description of the trailing edge flap for operation in the two modes is shown in Figure 30.

Figure 31 presents the effectiveness of a half span and full span trailing edge flap in terms of (1) moment of area of flap to control surface and (2) flap deflection to balance control deflection (measured relative to local airstream) at supersonic speeds. The flap size can be based on trim tab requirements while anti-balance effectiveness will depend on the gearing ratio provided (i.e. ratio of tab deflection to control deflection).

DRAG AT MACH 3.0

The drag consists of the zero-lift drag, induced drag and trim drag.

Induced Drag - Wing

The wing drag due to lift was obtained from the wing optimization program and is presented in Figure 21 as a function of wing C_L.

Induced Drag - Body

The drag due to lift of the body components is given by

$$C_{D_i} = C_N \sin \alpha'$$

where α' = angle of attack of component.

ZERO LIFT DRAG

A zero lift drag breakdown for the MC-10 is presented in Table IV.

Wave Drag - Wing

The wing thickness drag was obtained from the theoretical and experimental data of Reference 3.

The wing used in this study has a rounded leading edge with a diameter of 2 inches as dictated by aerodynamic heating. The leading edge drag is given by

$$C_D = 1.33 \frac{S_{L.E.}}{S_{ref}} \cos^3 \Lambda_{L.E.}$$

where $S_{L.E.}$ = leading edge cross sectional area.

The wing has a rounded base with a 3 in. diameter. The base drag is given by

$$C_D = \left(\frac{\Delta P}{q}\right)_{Base} \frac{S_B}{S_{ref}}$$

where S_B = effective base cross sectional area.

In order to simplify the analysis the 3 in. diameter rounded base was assumed to be equivalent to a 2.5 in. thick flat base. The 2.5 in. was used in finding the effective base cross-sectional area.

Wave Drag - Pylon

Due to the end plate effects of the wing and the body the pylon was essentially an infinite aspect ratio airfoil with a biconvex airfoil section. The drag coefficient is given by

$$C_D = \frac{5.33\,(t/c)^2}{\beta} \frac{S_P}{S_{ref}}$$

where 5.33 is shape constant from Reference 6. S_P = pylon planform area.

Wave Drag - Fin

The fin wave drag was obtained from the theoretical and experimental data of Reference 3.

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Wave Drag - Fuel Capsule

On configuration MC-10 and several other configurations a larger fuel volume than originally estimated was needed. This volume was obtained by thickening the pylon. The increase in drag due to this thickening was estimated by assuming this increased volume took the form of wedges on the sides of the original pylon. A drag coefficient was obtained using the wedge pressure coefficient data of Reference 4.

Wave Drag - Canopy and Fuel Capsule (Twin Engine Configurations)

The wave drag of the canopy and fuel capsule was obtained from shock expansion theory. The flow was calculated over the forebody as if it were a half cone and then a two dimensional expansion was used to calculate the flow properties over the afterbody. The base drag was calculated using the base pressure coefficients from Reference 5.

Engine Drag

The engine external drag coefficients were supplied by Marquardt and are shown in Figure 13. These drag coefficients are based on engine area A_3. The drag coefficient in terms of wing area is given by

$$C_D = C_D' \frac{A_3}{S_{ref}}$$

where C_D' = drag coefficient based on A_3.

Friction Drag

The friction drag was evaluated at Mach 3.0 and 135,000 ft. using the method of Frankl and Voishel where the friction drag coefficient is given by

$$C_{D_F} = \frac{.472}{(\log_{10} R_N)^{2.58} \left(1 + \frac{\gamma - 1}{\gamma} M^2\right)^{.467}} \quad \frac{S_{wetted}}{S_{ref}}$$

Trim Drag

The trim drag is the increment in drag due to lift resulting from control surface deflection.

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Lift to Drag Ratio, L/D

The value of L/D is given by

$$\frac{L}{D} = \frac{C_L}{C_D} = \frac{C_L}{C_{D_o} + C_{D_i}}$$

This configuration is designed to start cruise at near maximum L/D with no trim drag. Since the engine is a constant geometry engine and is sized on the basis of a constant thrust coefficient the total drag coefficient should remain the same throughout the flight. As the last 2/3 of the fuel are expended the c.g. moves forward and trim drag increases. Since the total drag coefficient must remain constant the drag due to lift of the configuration neglecting trim drag must decrease. This means the lift coefficient, and hence L/D, decreases. Figure 32 presents a plot of trim lift and drag characteristics for the single and twin engine configurations. Figure 33 presents a plot of trimmed L/D against lift coefficient for the basic MC-10 configuration.

DIRECTIONAL STATIC STABILITY

The directional stability characteristics were analyzed in order to obtain the vertical fin size required to provide a directionally stable configuration.

The directional characteristics were evaluated at an angle of attack of 0°. All interference effects between components were neglected. The body terms evaluated were the internal and external engine contributions.

The internal side force coefficients and centers of pressure were obtained as outlined in the lift section of this appendix with the exception that there was only one internal lift term since the engine did not have a bend in the planform view.

Reference 6 states that the normal force on a ducted cone or ogive can be approximated by the normal force on the equivalent extended solid body. The engines used in this study have basically a two cone frustrum cowl. One rather blunt cone at the inlet and a second more slender cone for the afterbody. The data data of References 7 and 8 were analyzed to obtain the side force and center of pressure of the external body.

The pylon was treated as a two dimensional airfoil due to the end plate effects of the wing and body.

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The fin characteristics were obtained using the experimental data of Reference 9. It was found that a total fin area equal to 14% of the wing area would provide sufficient static directional stability at Mach 3.0 with the aft c.g.

The subsonic directional stability was investigated briefly. The side force and yawing moment coefficient terms were calculated in the manner outlined above.

A plot of yawing moment coefficient slope against Mach number for the forward and aft c.g. positions is shown in Figure 34. This figure shows that the vehicle has static directional stability over the complete Mach number range.

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TABLE II

BASIC CHARACTERISTICS OF HAZEL CONFIGURATIONS

CONFIGURATION	MC-10	MC-11	MC-14	MC-15	MC-17	MC-19	MC-20	MC-22	MC-24	PC-20	PC-22	PC-24
Range - Nautical miles	3,200	3,200	4,000	3,200	3,200	3,200	3,200	3,200	3,200	3,200	3,200	4,000
Gross Weight - Lbs. (start of cruise)	13,800	12,400	27,000	12,200	15,415	18,310	13,145	8,600	16,500	14,350	9,700	13,990
Wing Area - Ft.^{2}	1,985	1,985	3,000	1,985	1,985	1,985	1,985	1,985	1,985	1,985	1,985	1,985
Number of Engines	1	1	1	1	1	1	2	2	2	2	2	2
Type of Fuel	PB	PB	PB	PB	PB	PB	PB	SF-1	PB	PB	SF-1	SF-1
Wing Loading - Start Cruise lb/Ft.^{2}	6.95	6.25	8.65	6.15	7.77	9.22	6.67	4.34	8.3	7.2	4.89	7.04
Start Cruise Alt.	125,000	128,000	125,000	128,300	125,000	125,000	125,000	139,000	125,000	125,000	136,200	125,000
End Cruise Alt.	137,800	143,400	141,200	141,000	136,800	137,400	137,200	146,000	140,100	136,400	141,800	135,000
Construction	Fabric	Fabric	Fabric	Fabric	Fabric	Plastic	Fabric	Fabric	Fabric	Fabric	Fabric	Fabric
AERODYNAMICS												
C_{D_0}	.01855	.0178	.0195	.01855	.01885	.02044	.01862	.02013	.0192	.01782	.0164	.0220
L/D) Start Cruise	4.17	4.25	4.15	4.18	4.21	4.11	4.26	4.18	4.17	4.47	4.24	3.55
L/D) End Cruise	3.66	3.7	3.5	3.66	3.6	3.48	3.72	3.57	3.61	3.81	3.61	3.05
α Start Cruise	8.5°	8.5°	10.1°	8.5°	9.15°	10.3°	8.0°	8.6°	9.15°	8.4°	8.6°	8.25°
α End of Cruise	9.2°	9.2°	10.6°	9.2°	9.7°	10.8°	8.5°	9.0°	9.7°	8.8°	9.0°	8.7°
ENGINE												
Capture Area - Ft.^{2}	85.3	83.6	167	85.6	94	115	40 per eng.	39.2 per eng.	51.0 per eng.	2[illegible].7 per eng.	30 per eng.	40.8 per eng.
C_T (or Thrust)	.85	.85	.85	.85	.85	.85	.85	1.0	.85	1,580 lbs.	1,191 lbs.	1,970 lbs.
SFC at start cruise	1.458	1.489	1.458	1.49	1.458	1.458	1.458	.98	1.7	1.95	.92	.865
SFC at end cruise	1.586	1.658	1.625	1.625	1.575	1.58	1.58	1.04	2.08	2.03	.958	.912
Manufacturer	Marquardt	Marquardt	Marquardt	Marquardt	Marquardt	Marquardt	Marquardt	Marquardt	Marquardt	P and W	P and W	P and W
WEIGHTS - LBS.												
Payload	800	800	800	300	1,300	800	800	800	800	800	800	800
Crew	200	200	200	200	200	200	200	200	200	200	200	200
Structure	2,635	2,397	5,062	2,290	2,900	4,316	2,508	1,623	2,978	2,393	1,796	2,320
Fixed Equipment	2,307	2,225	3,175	2,342	2,347	2,376	2,272	2,107	2,534	2,379	2,124	2,552
Miscellaneous	68	65	103	68	68	68	100	100	100	100	100	100
Engine	1,460	1,200	2,960	1,350	1,600	1,800	1,340	920	1,338	1,470	1,710	2,198
Glide Weight	7,470	6,887	12,300	6,550	8,415	9,560	7,220	5,750	7,950	7,342	6,730	8,170
Fuel Weight	6,330	5,513	14,700	5,650	7,000	8,750	5,925	2,850	8,550	7,008	2,970	5,820
Gross Weight (start of cruise)	13,800	12,400	27,000	12,200	15,415	18,310	13,145	8,600	16,500	14,350	9,700	13,990

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TABLE III

BASIC CHARACTERISTICS OF ROCKET POWERED HAZEL CONFIGURATIONS

	Boost-Glide	Boost-Rocket Cruise-Glide
Range, N. Miles	3,200	3,200
Maximum Velocity, Ft/Sec	14,500	8,800
Mid Range Altitude, Ft.	177,000	150,000
Average Aerodynamic L/D	4.7	5.0
Landing Speed, Knots	50	55
Weight, Lbs.		
Start Glide	4,000	4,750
Cruise Fuel	-	7,450
Start Cruise	-	12,200
3rd Stage Motor	420	-
3rd Stage Fuel	4,200	-
Start of 3rd Stage	8,620	-
2nd Stage Motor	905	1,110
2nd Stage Fuel	9,050	11,100
Start of 2nd Stage	18,575	24,410
1st Stage Motor	1,950	2,225
1st Stage Fuel	19,500	22,250
Launch Wt.	40,025	48,885

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TABLE IV

MC-10

ZERO-LIFT DRAG BREAKDOWN

M=3.0 135,000 Feet

Wave Drag	
Wing Thickness	.00334
Wing Base	.000875
Wing Leading Edge	.00176
Fin	.00030
Pylon	.00126
Engine	.00440
Fuel Capsule	.00055
Total Wave	.012485
Friction Drag	
Wing	.00481
Fin	.00090
Pylon and Fuel Capsule	.00035
Total Friction	.00606

Total Wave Drag	.012485
Total Friction Drag	.00606
Total C_{D_o}	.018545

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CONFIGURATION MC-10 SECRET

FIGURE 1

Wt. Start of Cruise	13,800 lb.
Fuel Wt.	6,330 lb.
Wing Area	1,985 ft.^{2}
Fuel Type	Pentaborane

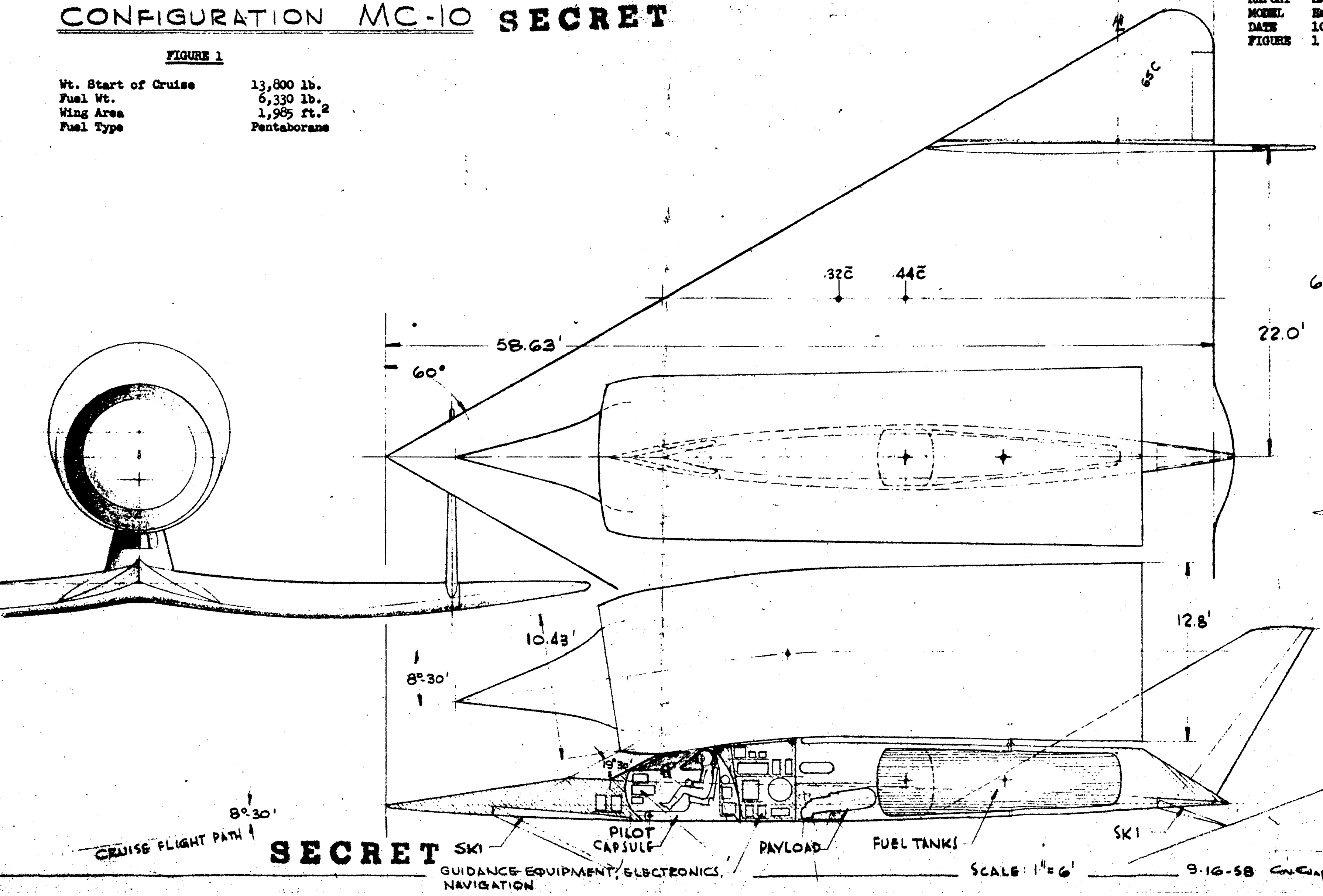

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FIGURE	2

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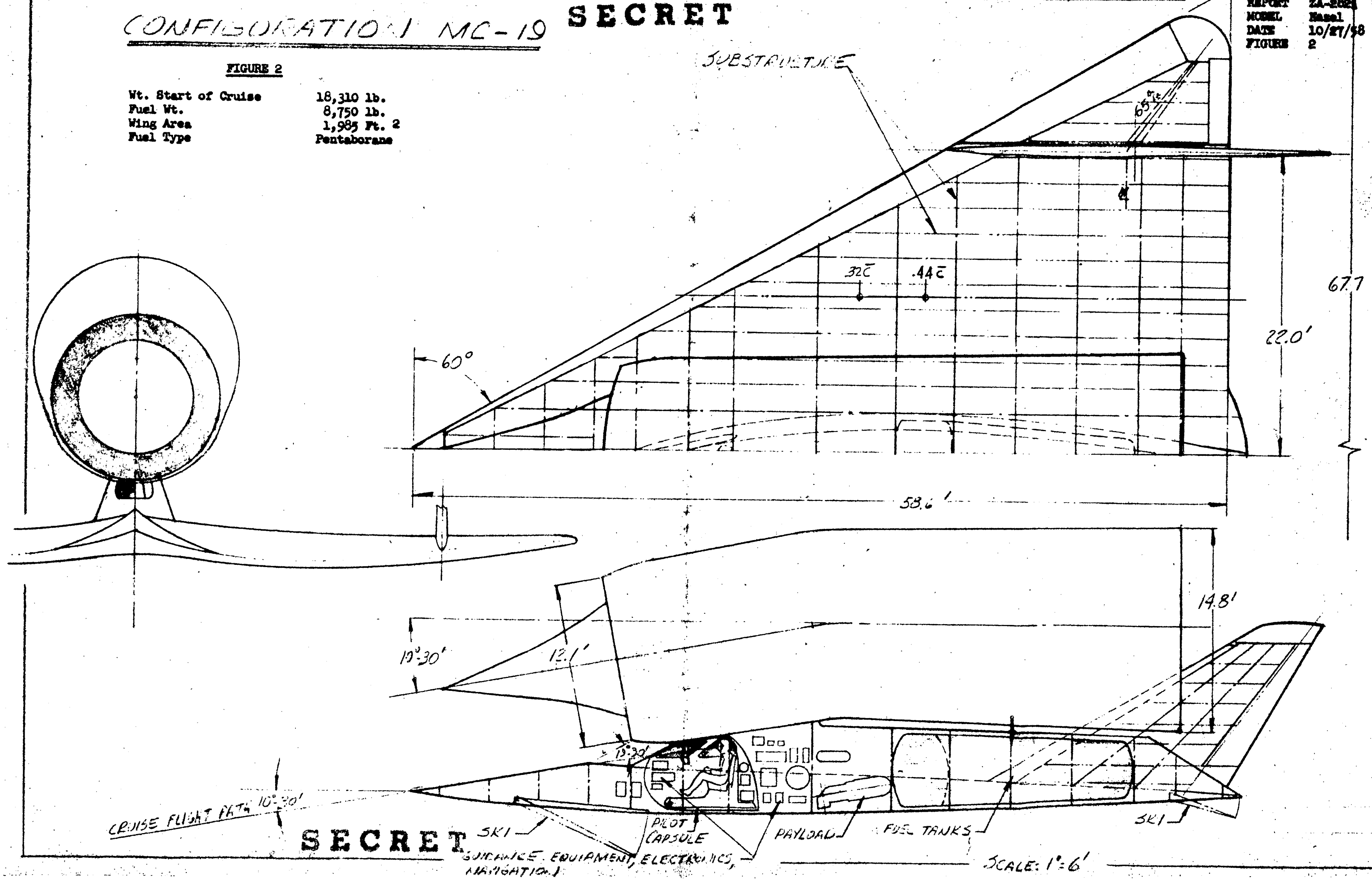
CONFIGURATION MC-19

FIGURE 2

Wt. Start of Cruise	18,310 lb.
Fuel Wt.	8,750 lb.
Wing Area	1,985 Ft. 2
Fuel Type	Pentaborane



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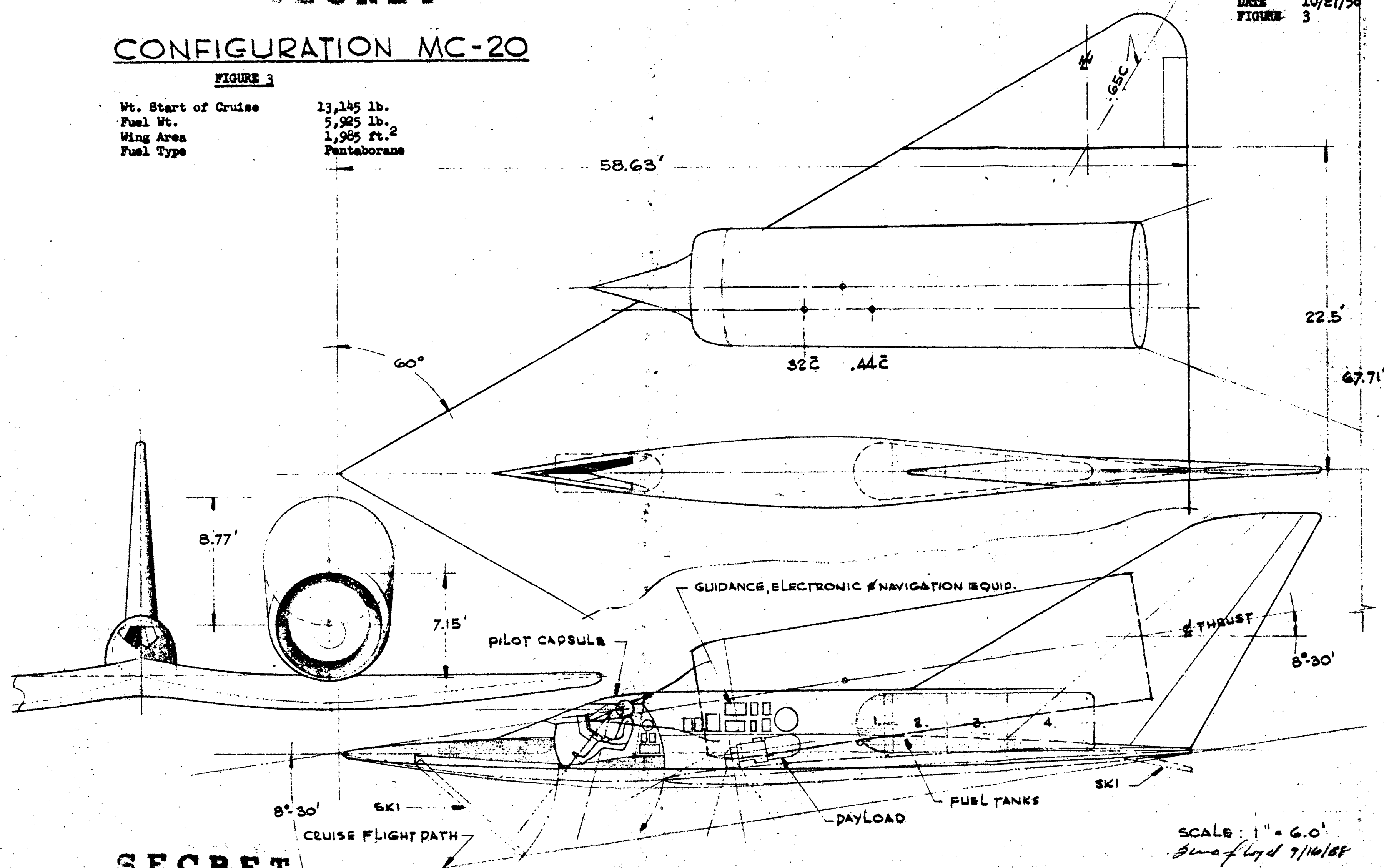
CONFIGURATION MC-20

FIGURE 3

Wt. Start of Cruise	13,145 lb.
Fuel Wt.	5,925 lb.
Wing Area	1,985 ft.^{2}
Fuel Type	Pentaborane



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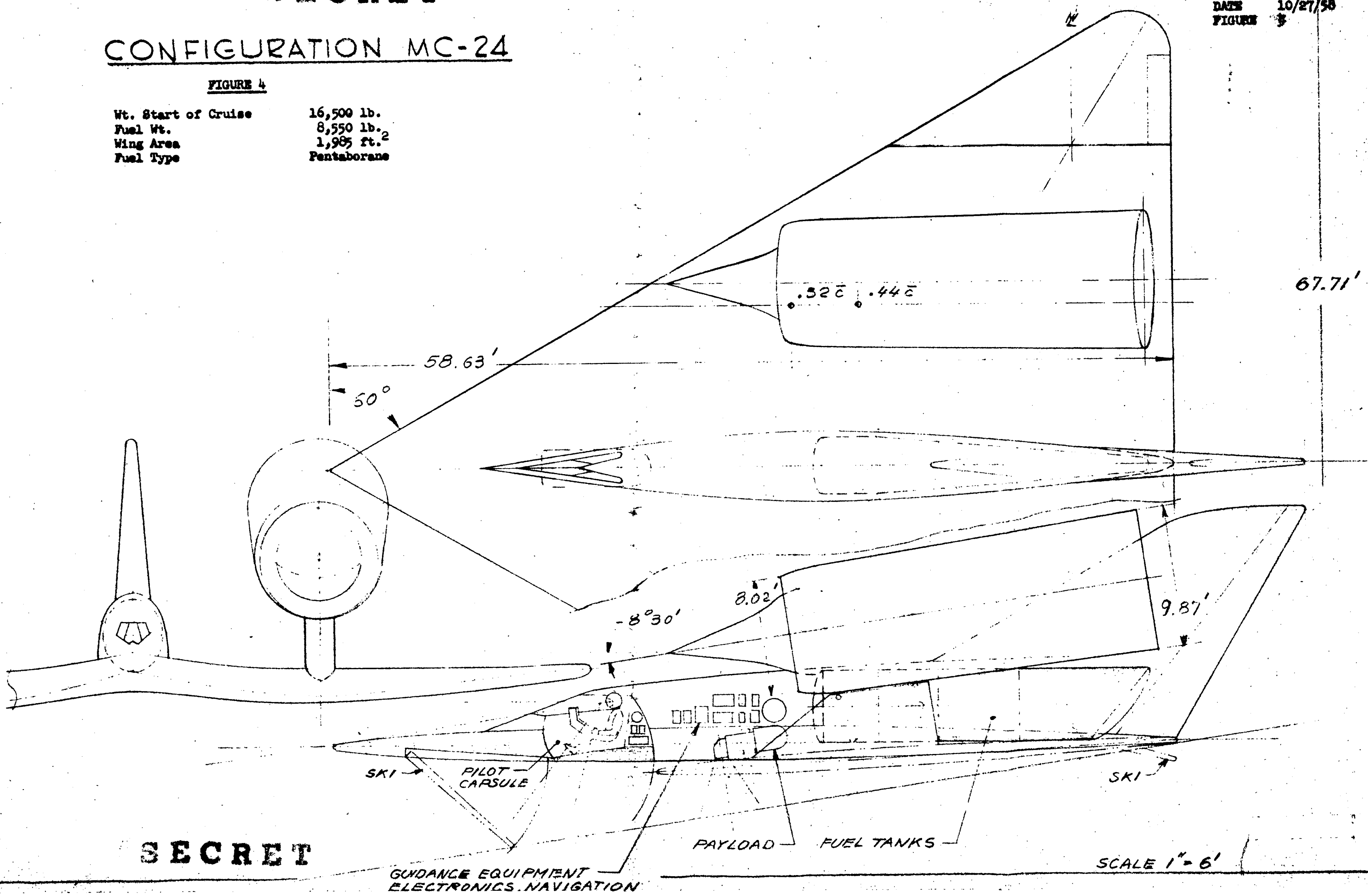
CONFIGURATION MC-24

FIGURE 4

Wt. Start of Cruise	16,500 lb.
Fuel Wt.	8,550 lb.
Wing Area	1,985 ft.^{2}
Fuel Type	Pentaborane



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CONFIGURATION PC-22

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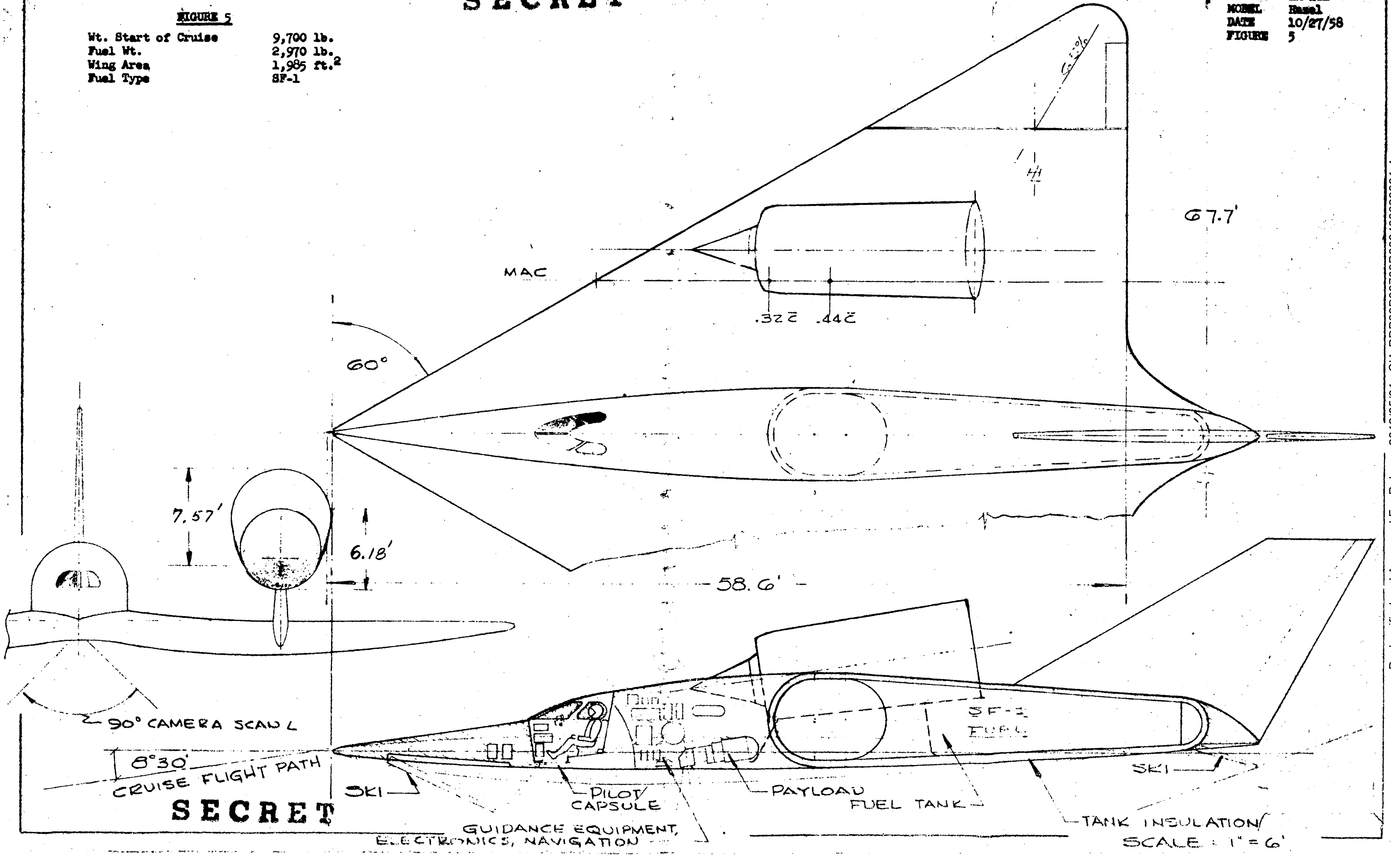
FIGURE	5

FIGURE 5

Wt. Start of Cruise	9,700 lb.
Fuel Wt.	2,970 lb.
Wing Area	1,985 ft.^{2}
Fuel Type	SF-1



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CONFIGURATION PC-24

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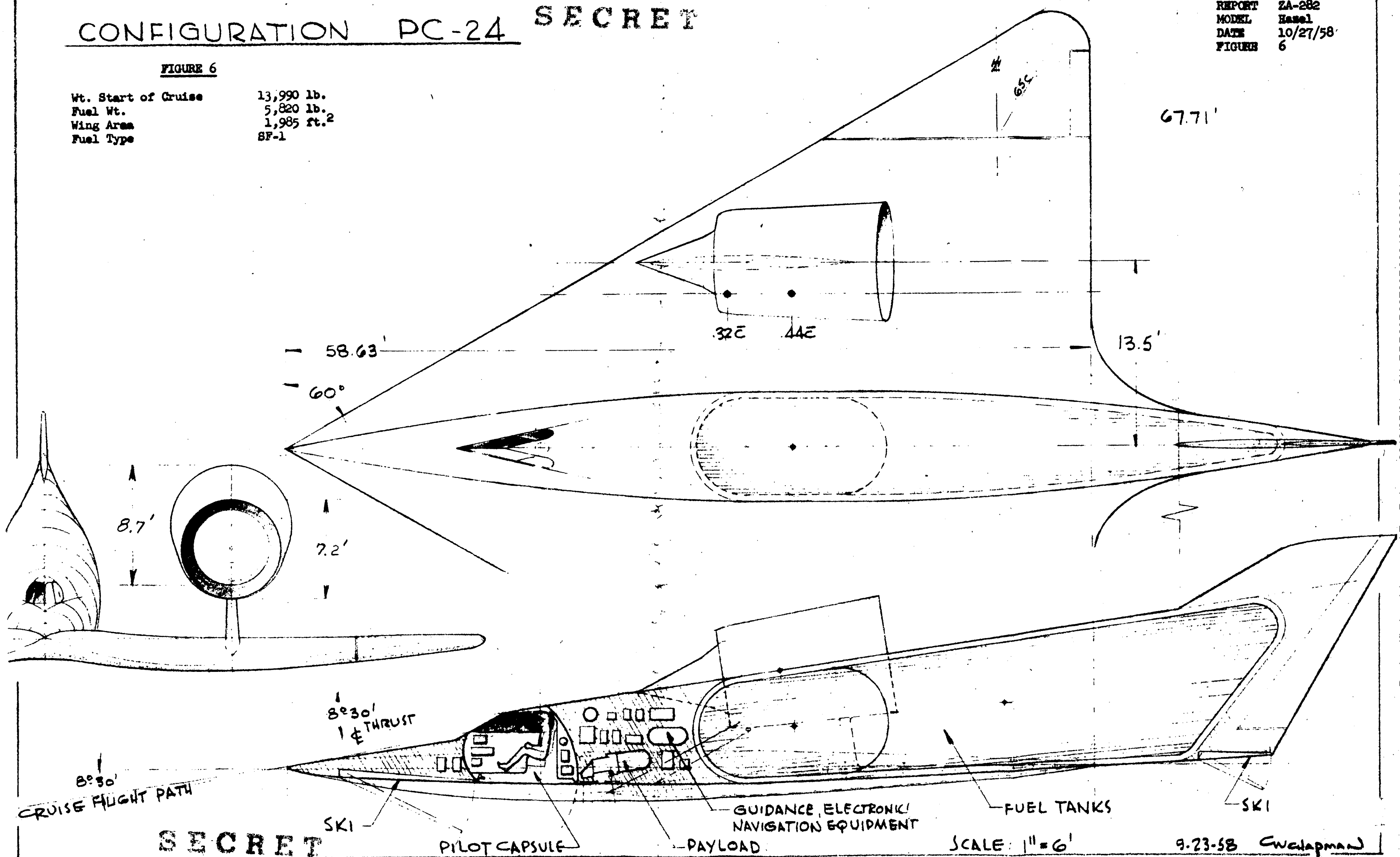
FIGURE	6

FIGURE 6

Wt. Start of Cruise	13,990 lb.
Fuel Wt.	5,820 lb.
Wing Area	1,985 ft.^{2}
Fuel Type	SF-1



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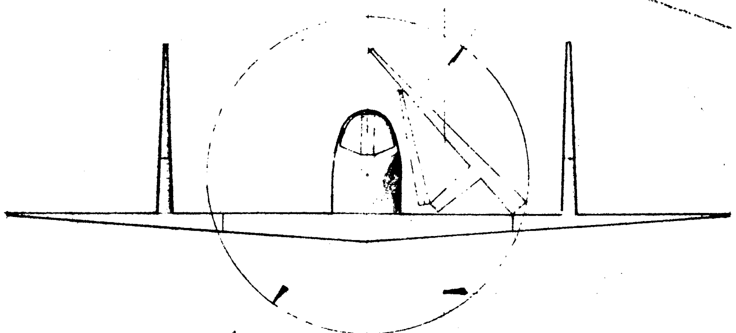
ROCKET-CRUISE CONFIGURATION HCG-2

(M8 @ 150,000') RFNA-HYDRAZINE

FIGURE 7

LAUNCH WT	48885 LB.
START OF CRUISE WT	12,200 LB
FUEL WT	7455 LB
WING AREA	575 SQ.FT.
W/S (MID-CRUISE)	15 LB/SQ.FT
W/S (GLIDE)	8.25 LB/SQ.FT.



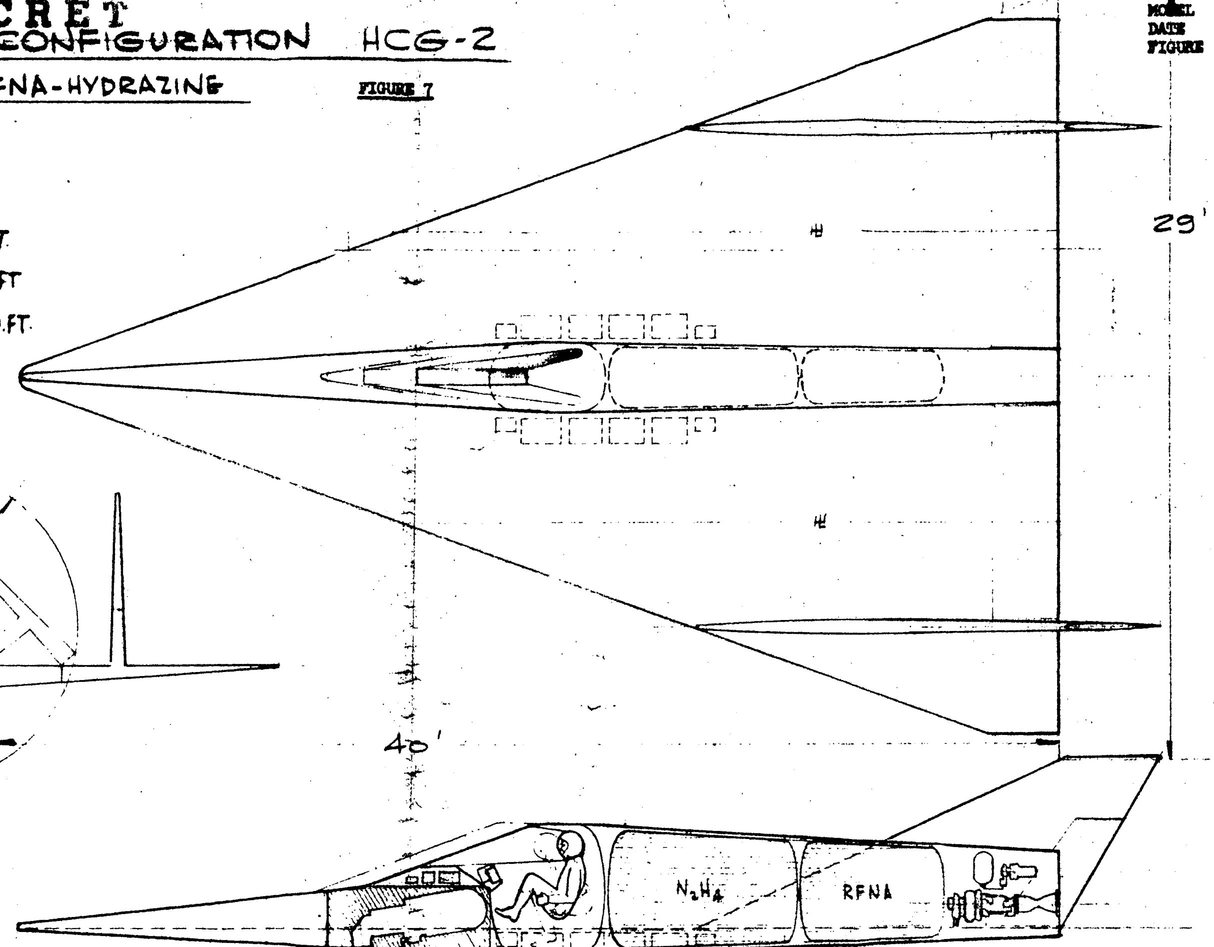


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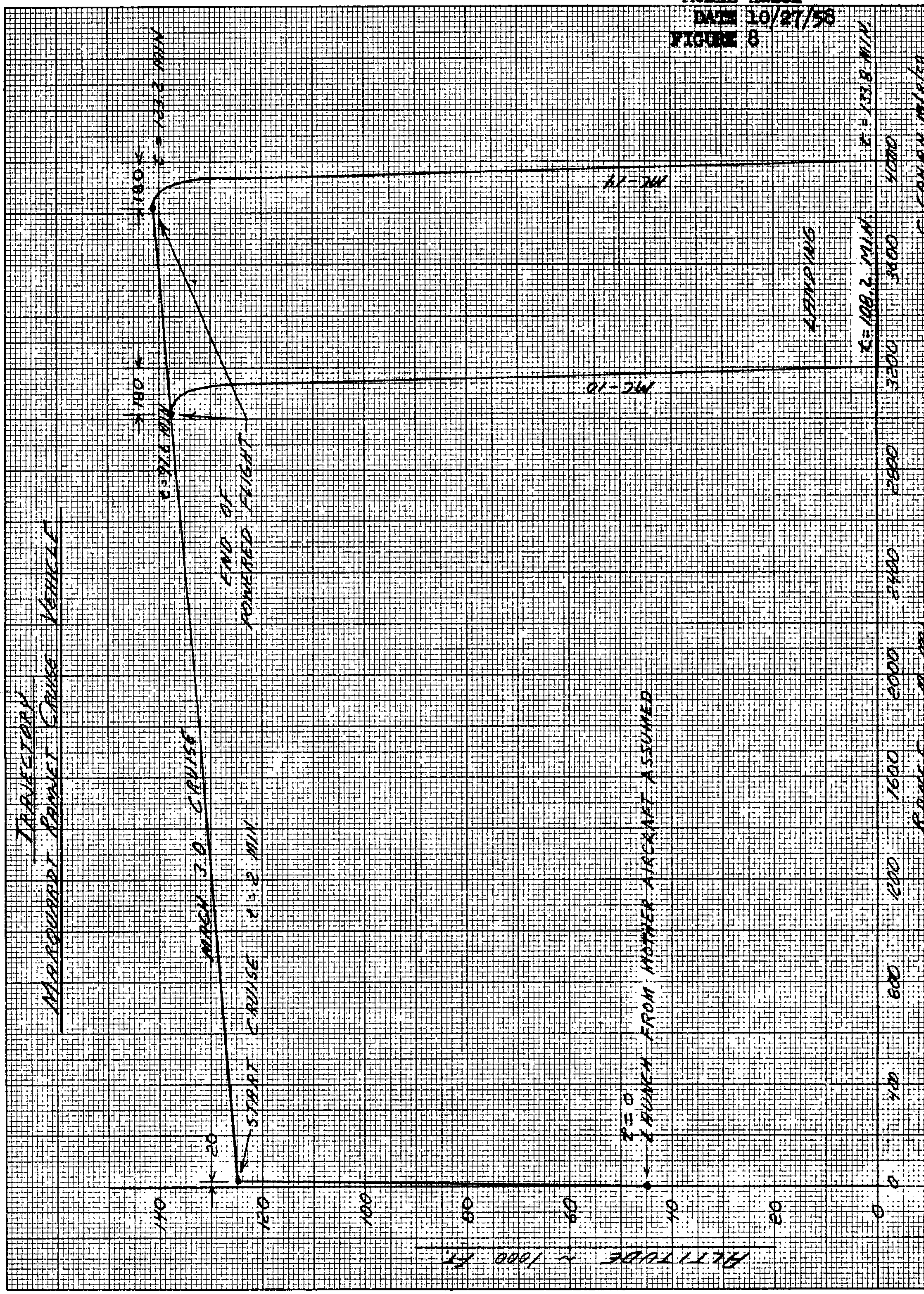
FIGURE 8



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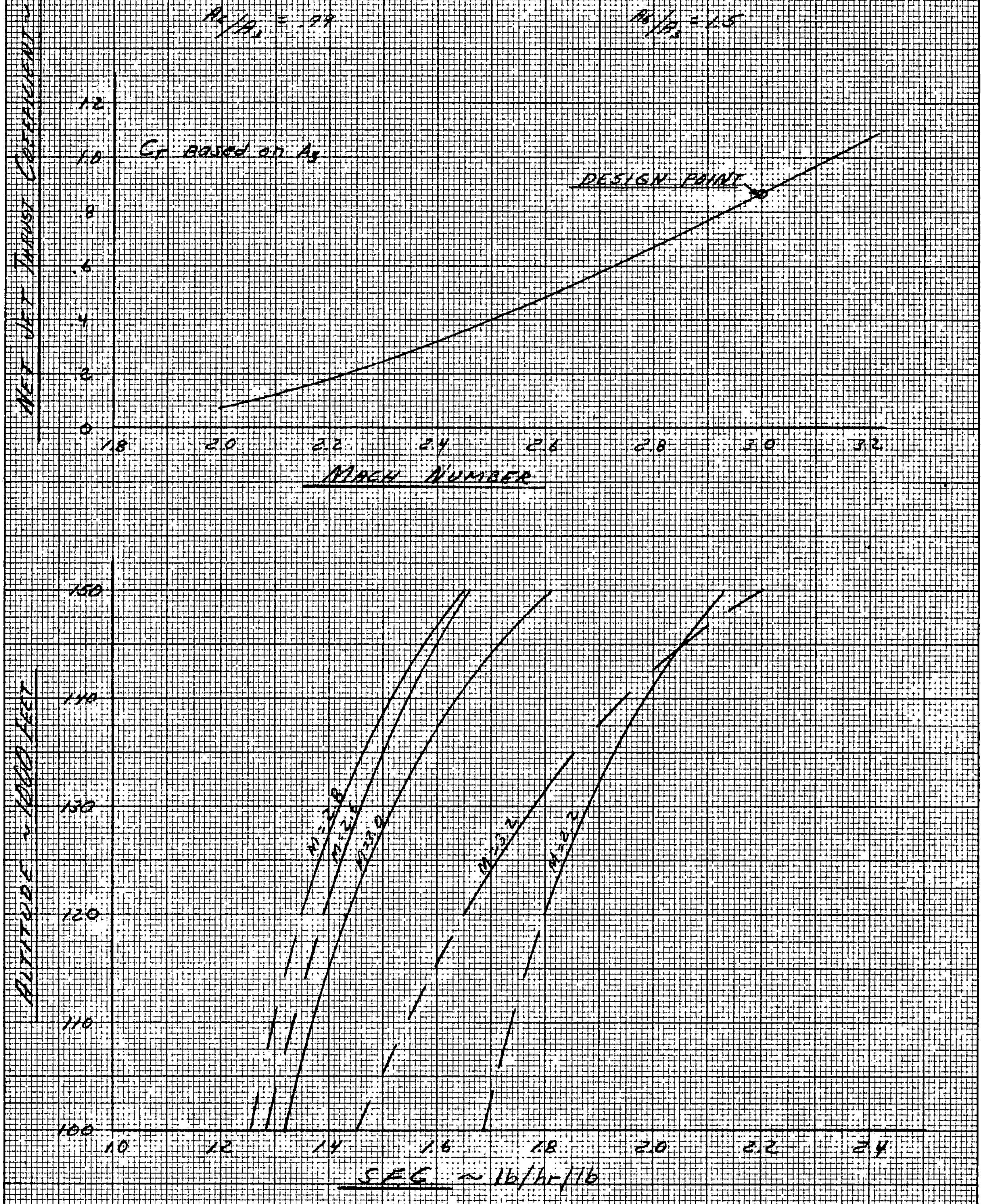
MARQUARDT RAMJET ENGINE CHARACTERISTICS

PENTABORANE FUEL



C. Cowan 10/9/58

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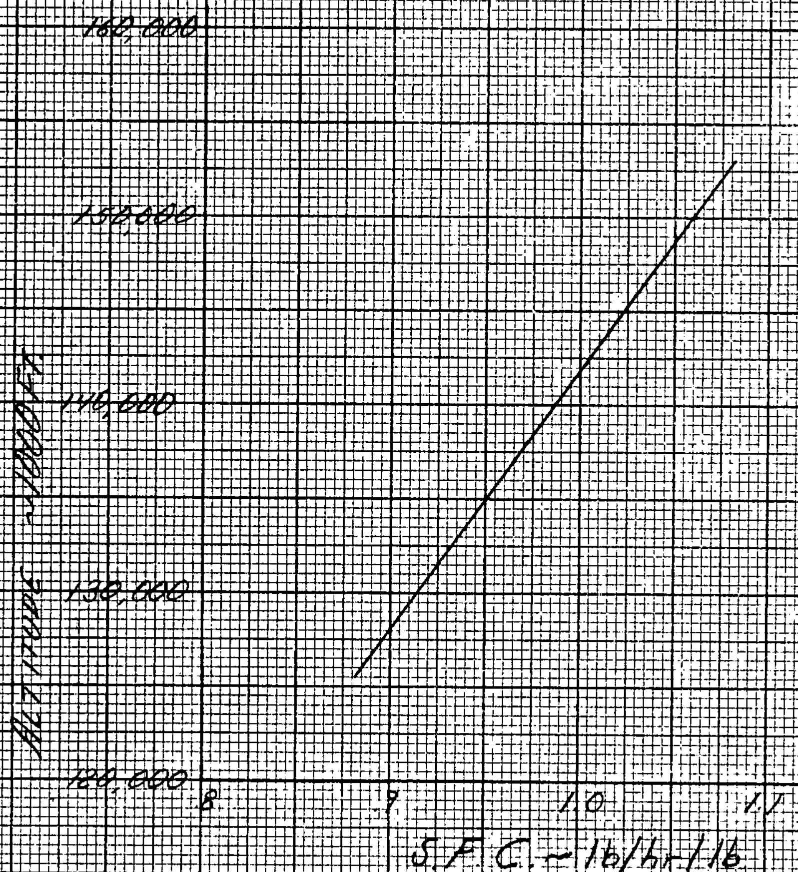
FIGURE 10

MARQUARDT RAMJET ENGINE CHARACTERISTICS

SF-1 FUEL M-3.0 A_C/A_3 = .99 A_6/A_3 = 1.5

C_T = 1.0



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MODEL Ramjet
DATE 10/27/58
FIGURE 11

MARQUARDT RAMJET
OFF-DESIGN PERFORMANCE

C_T vs M
Constant SFC Lines

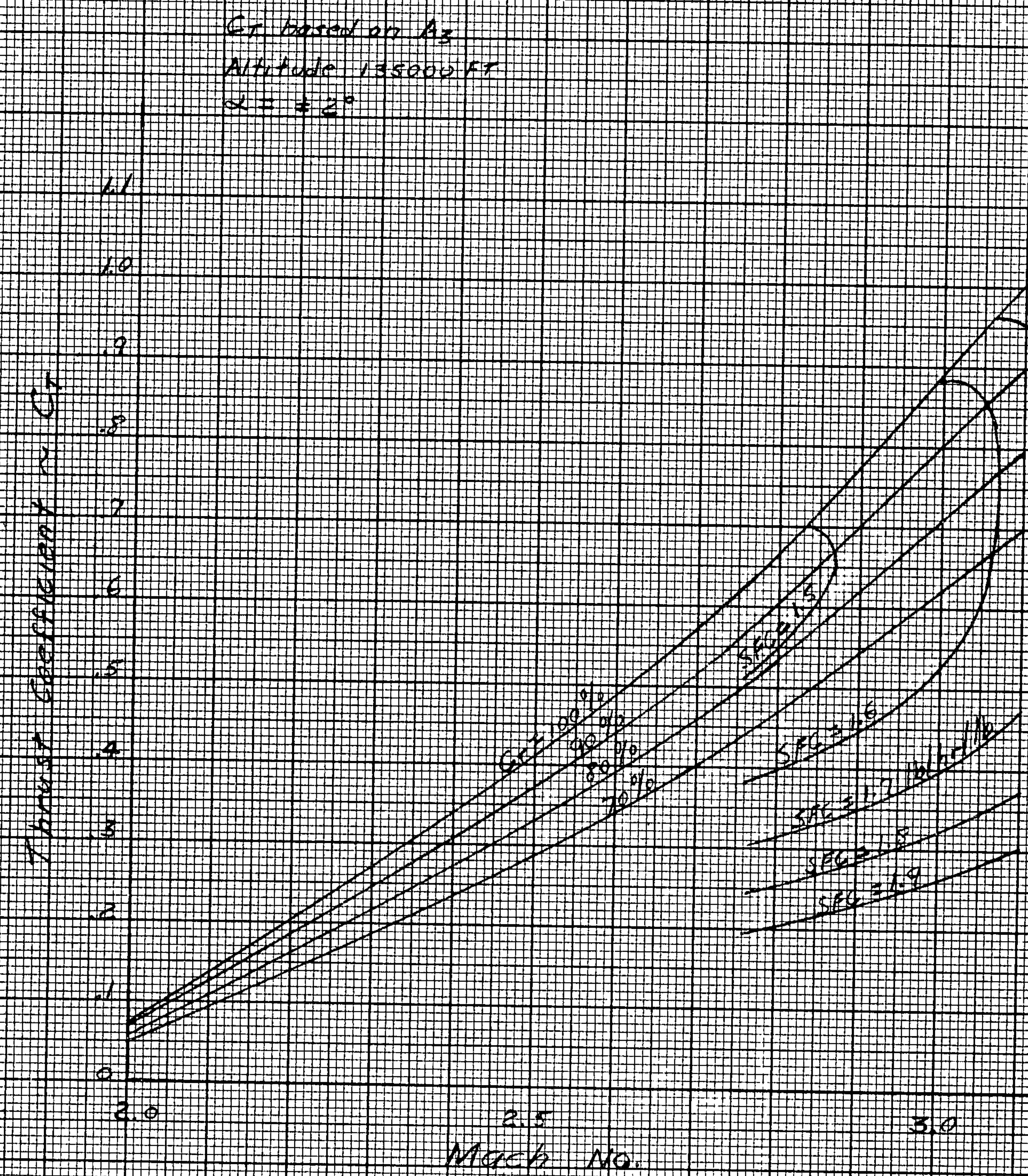

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FIGURE 12

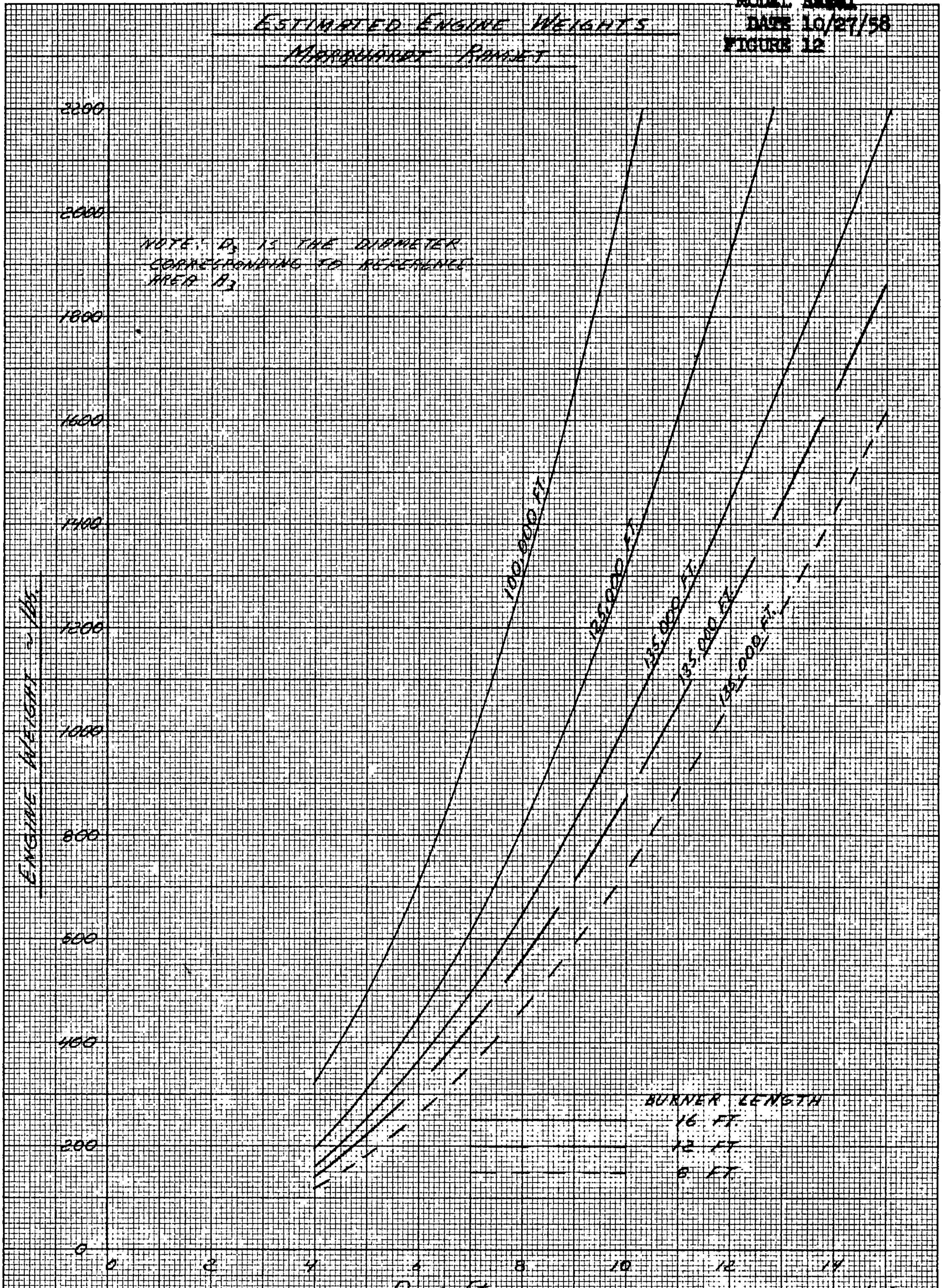

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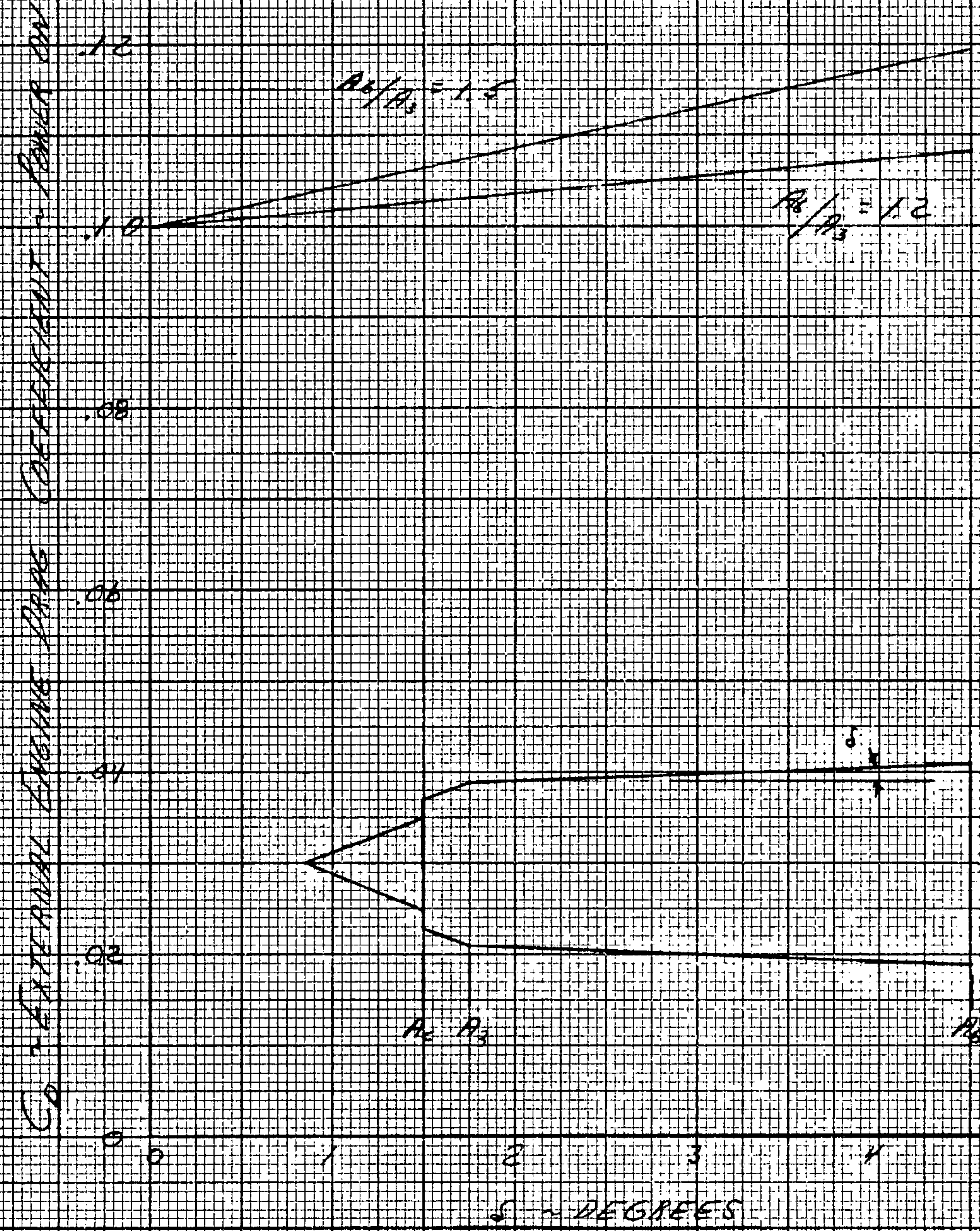
FIGURE 13

MARQUARDT RAMJET
POWER-ON EXTERNAL DRAG COEFFICIENT - C_D
$A_E/A_3 = .9$ $M = 3.0$

$C_D = \frac{D}{q A_3}$



C. Rohan 10/23/58

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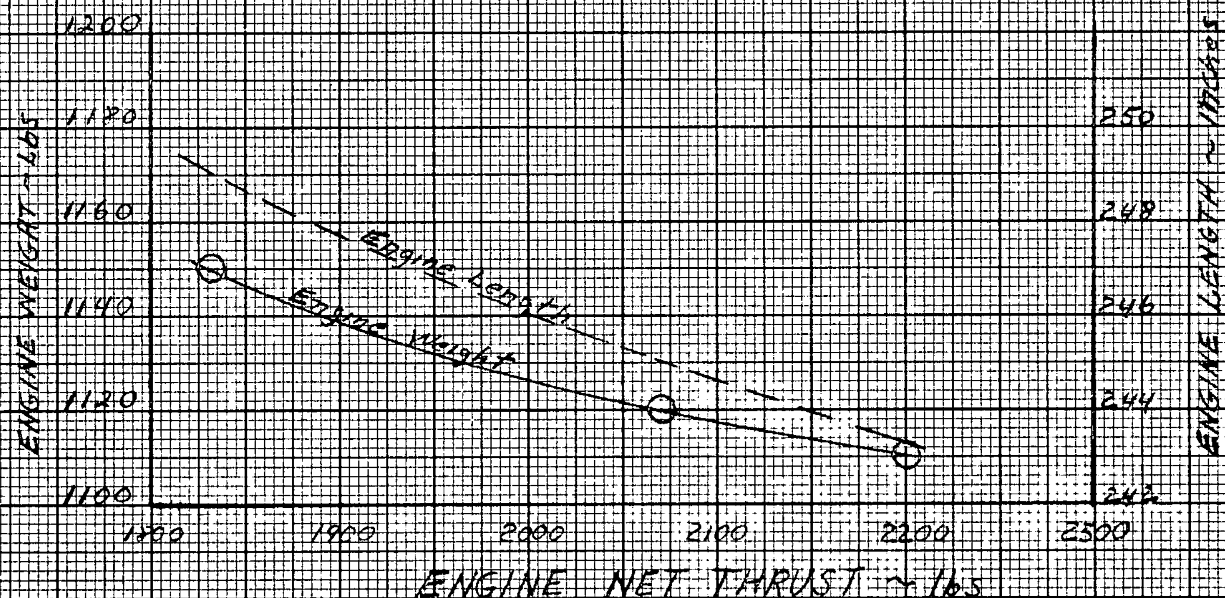
PRATT & WHITNEY RAMJET ENGINE
M=3.0
NET THRUST vs ENGINE WEIGHT
PENTABORANE FUEL

INLET DIAM. = 86.5"   EXIT DIAM. = 104"



VARIATION OF SFC WITH ALTITUDE

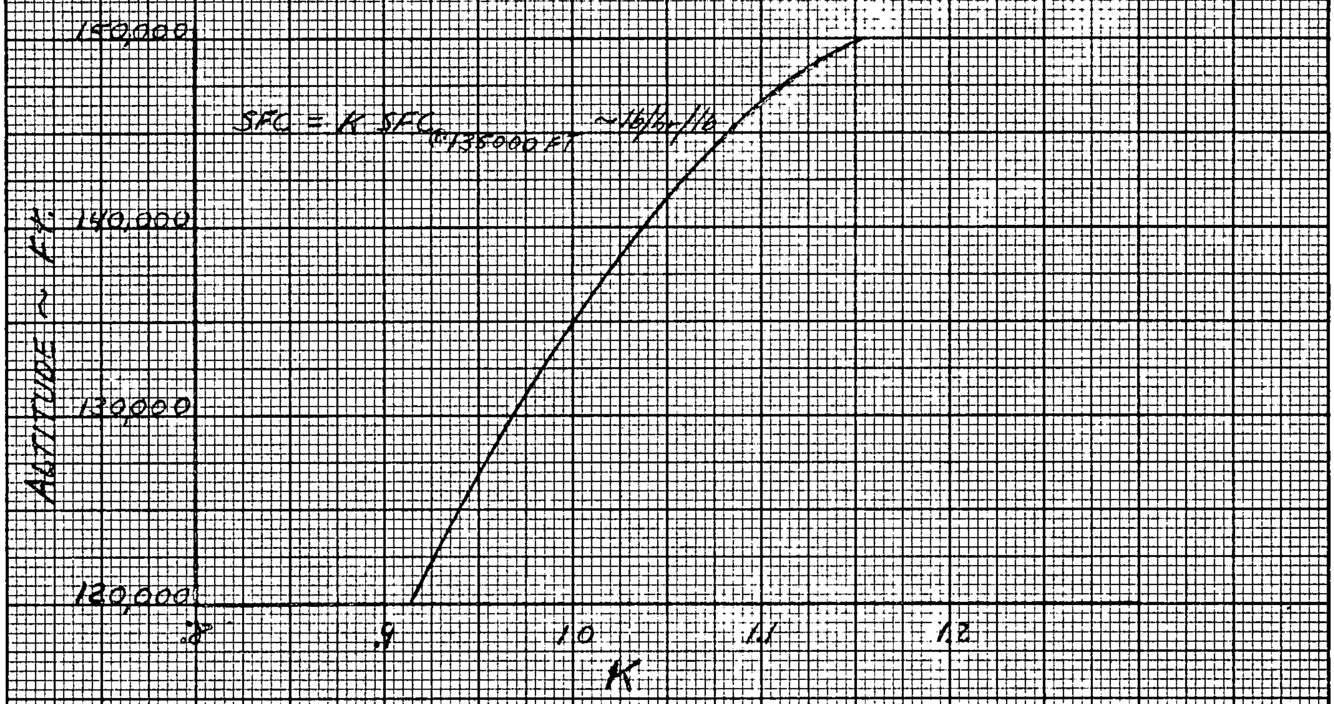


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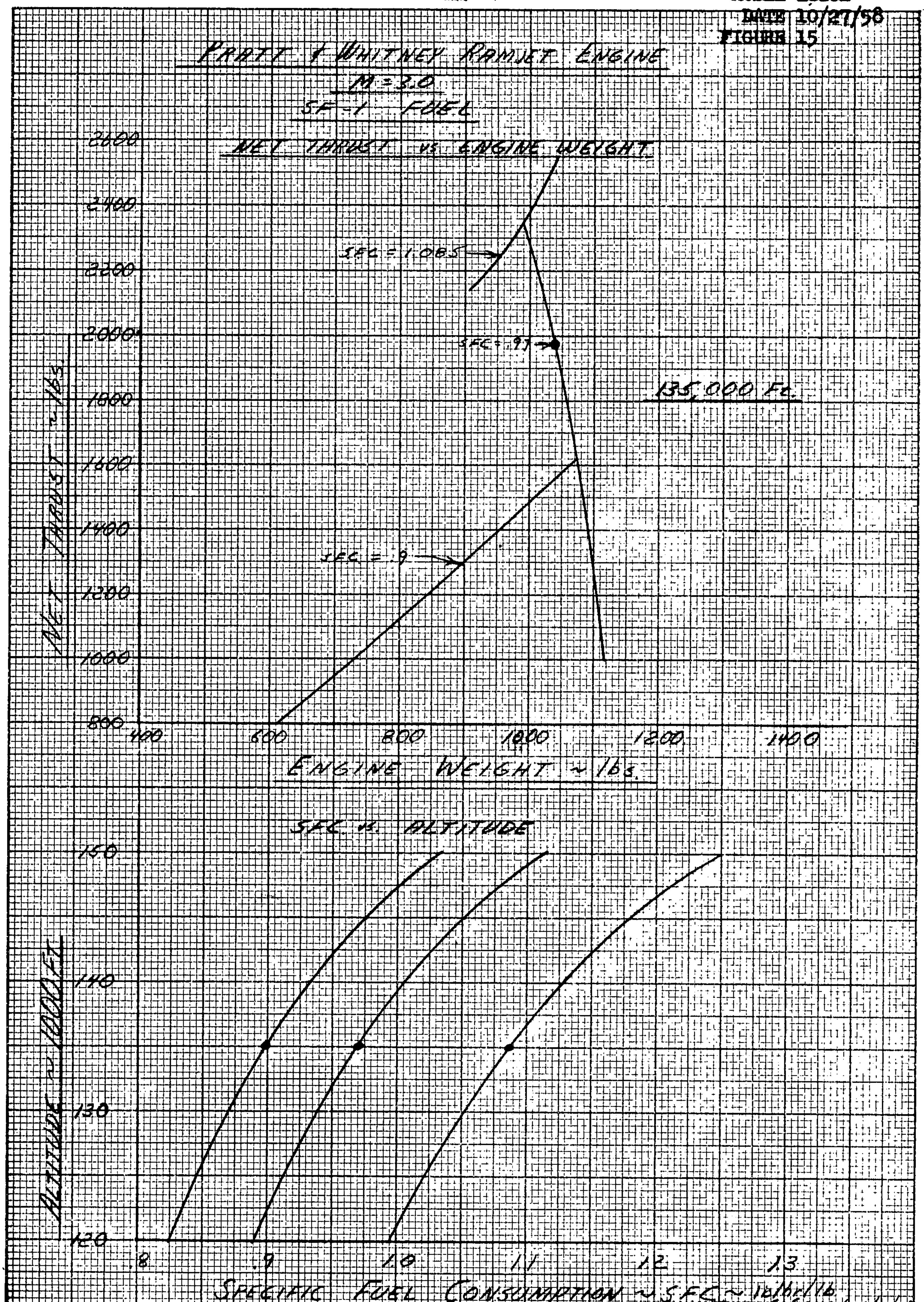
FIGURE 15



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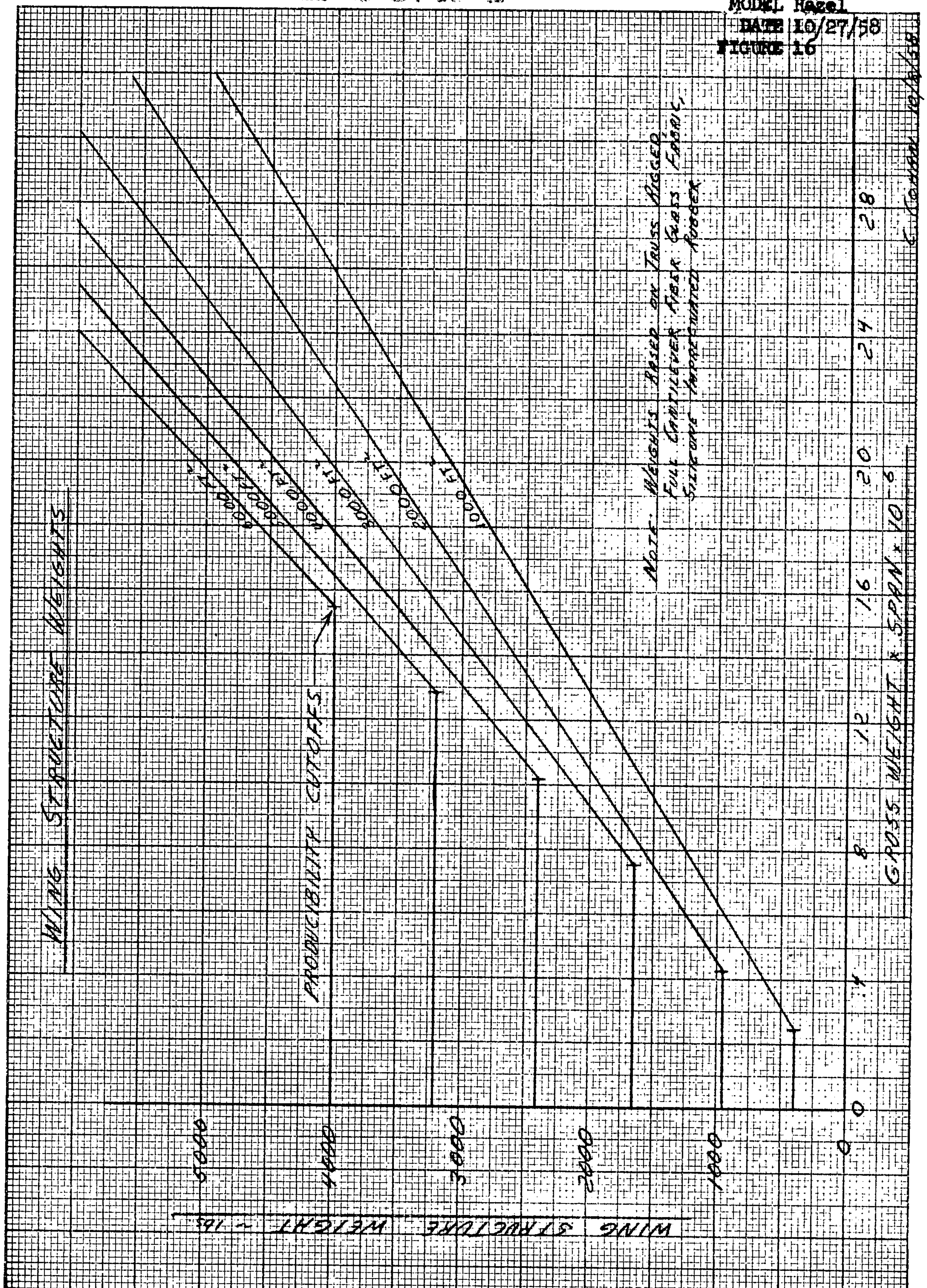
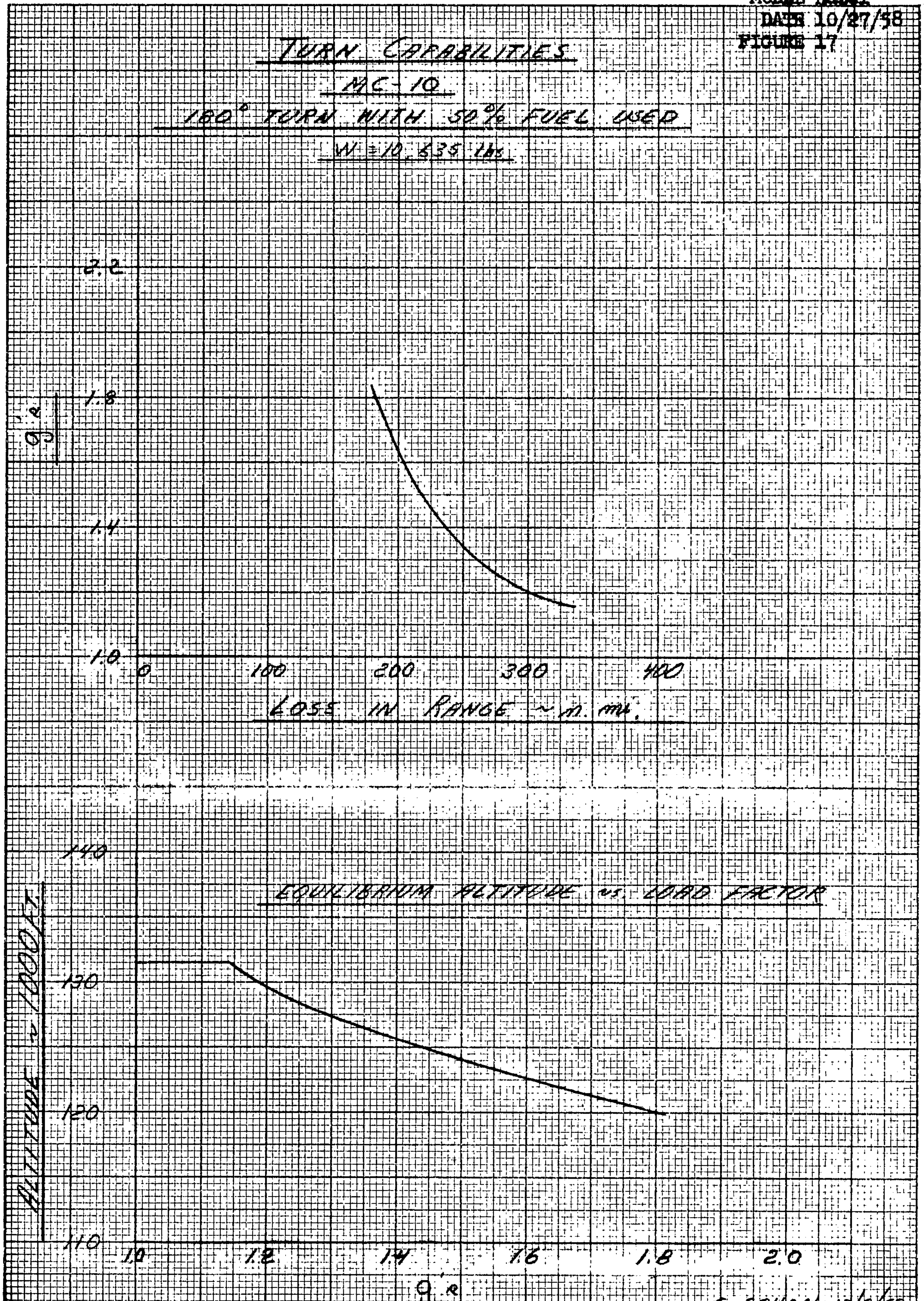
DATE 10/27/58
FIGURE 16



SECRET


TURN CAPABILITIES
MC-10
180° TURN WITH 50% FUEL USED
W = 10,635 lbs
2.2
1.8
1.4
1.0
g_R
0
100
200
300
400
LOSS IN RANGE ~ n. mi.
140
130
120
110
EQUILIBRIUM ALTITUDE vs. LOAD FACTOR
ALTITUDE ~ 1000 FT.
1.0
1.2
1.4
1.6
1.8
2.0
g_R
C. COHAN 10/9/58

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MODEL Hazel
DATE 10/27/58
FIGURE 18

SECRET

EFFECT OF CHANGES IN FIXED WT ON VEHICLE GROSS WT.

SINGLE ENGINE MARQUARDT
START CRUISE AT M=3.0 AT 125000 FT
20 N.MI BOOST + 3000 N.MI CRUISE + 180 N.MI GLIDE
= 3200 N.MI TOTAL RANGE

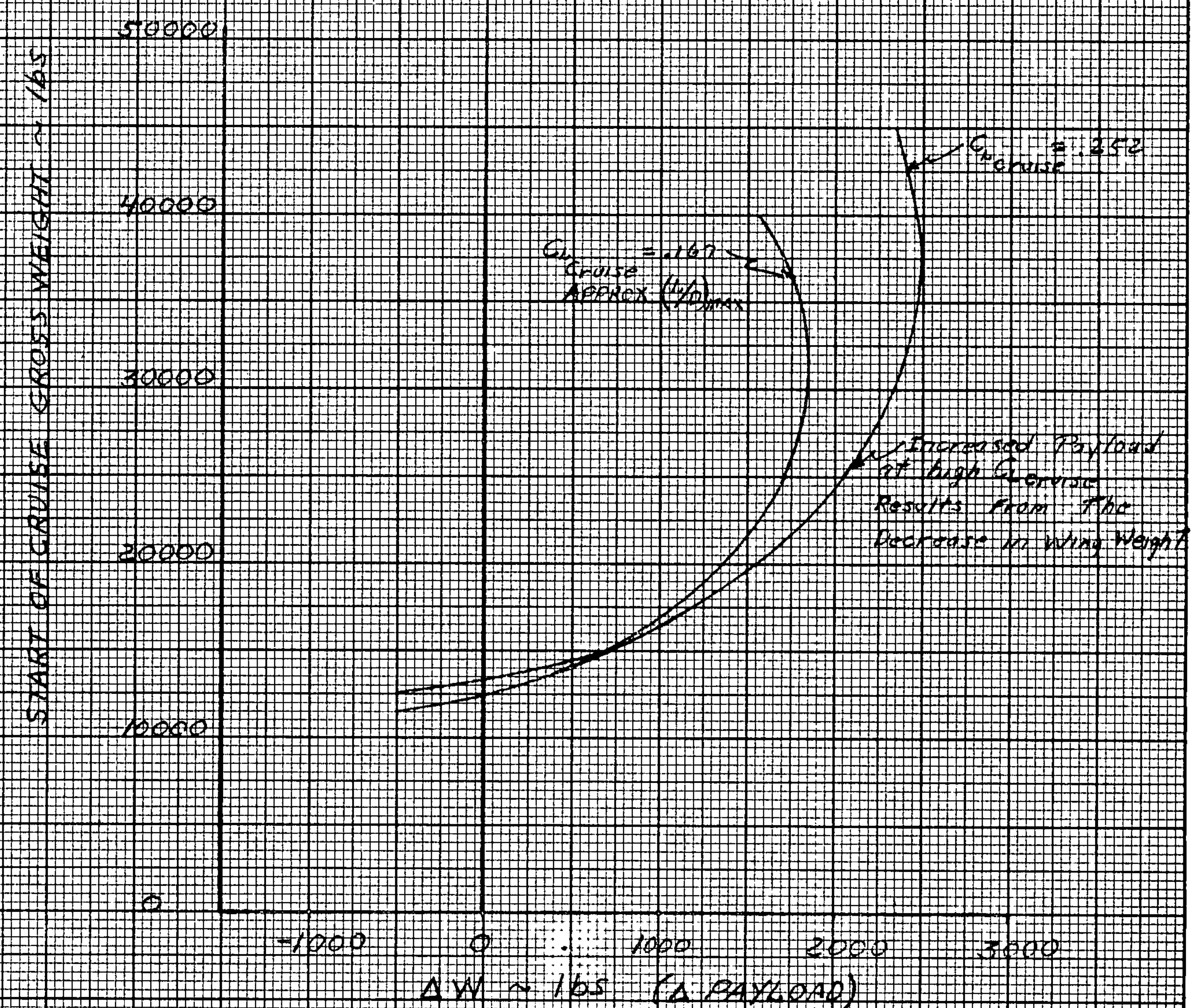

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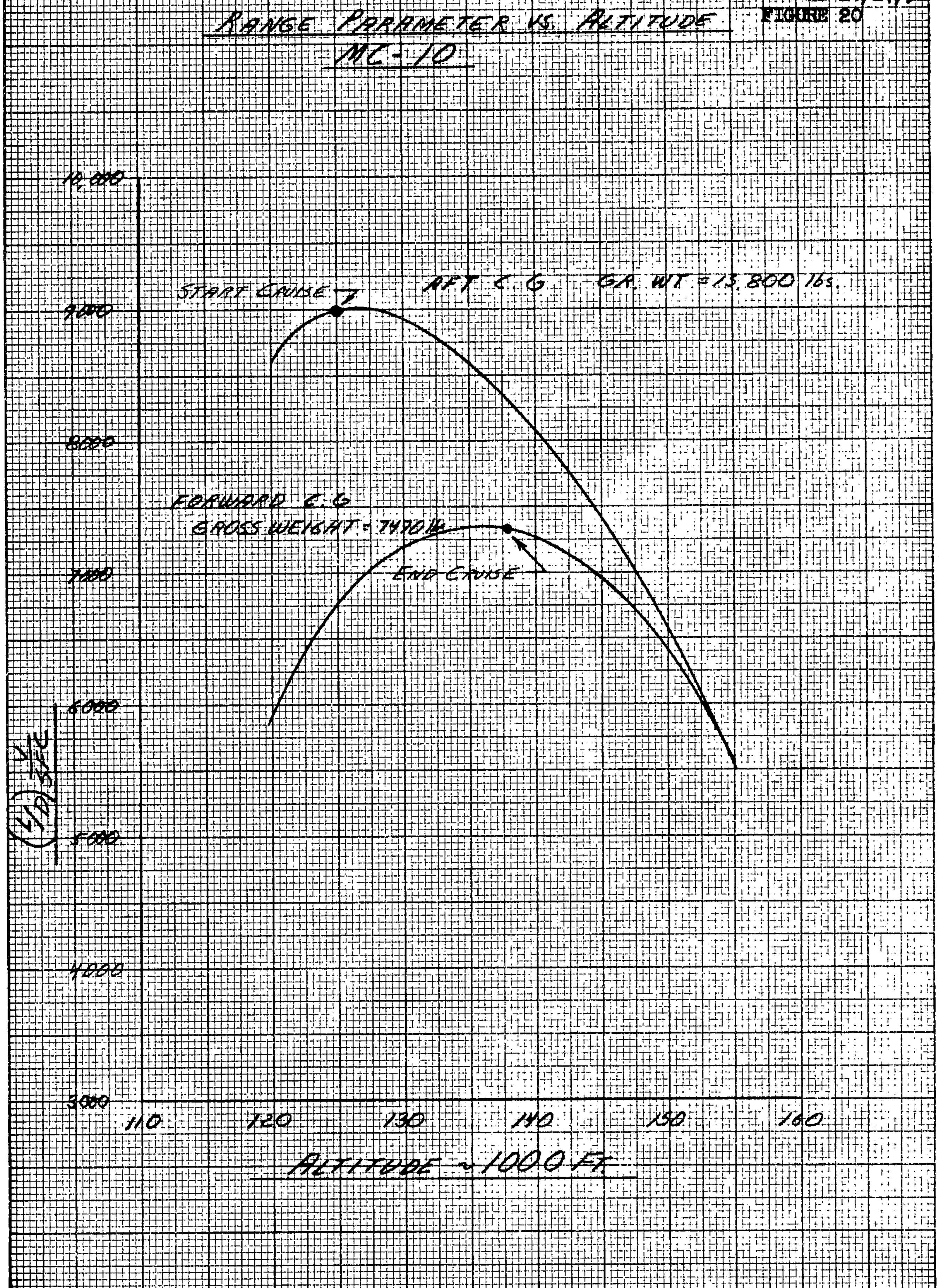
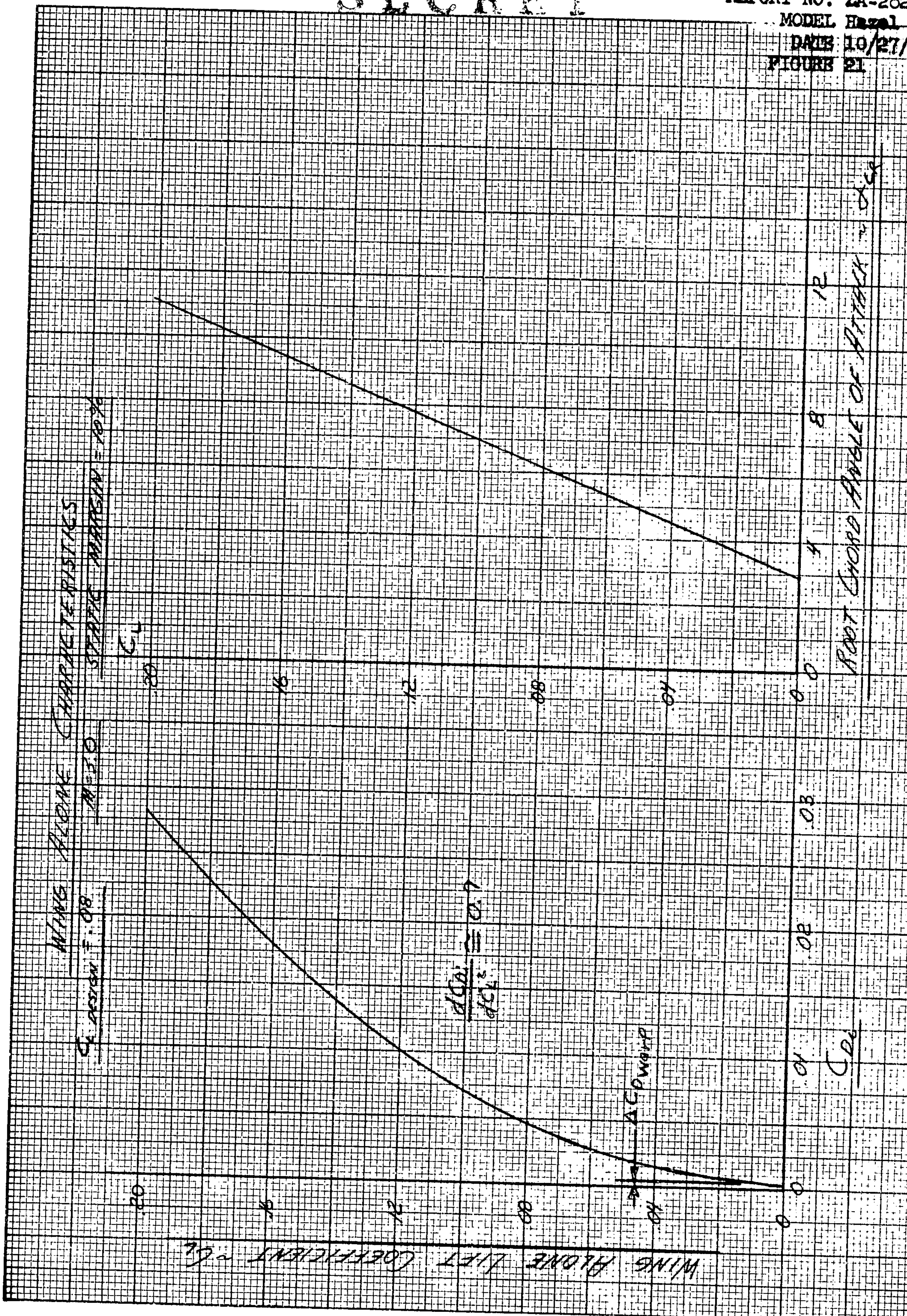
FIGURE 20



KEUFFEL & ESSER CO. 10 X 10 TO THE ½ INCH 359T-11G MADE IN U.S.A.

FIGURE 21



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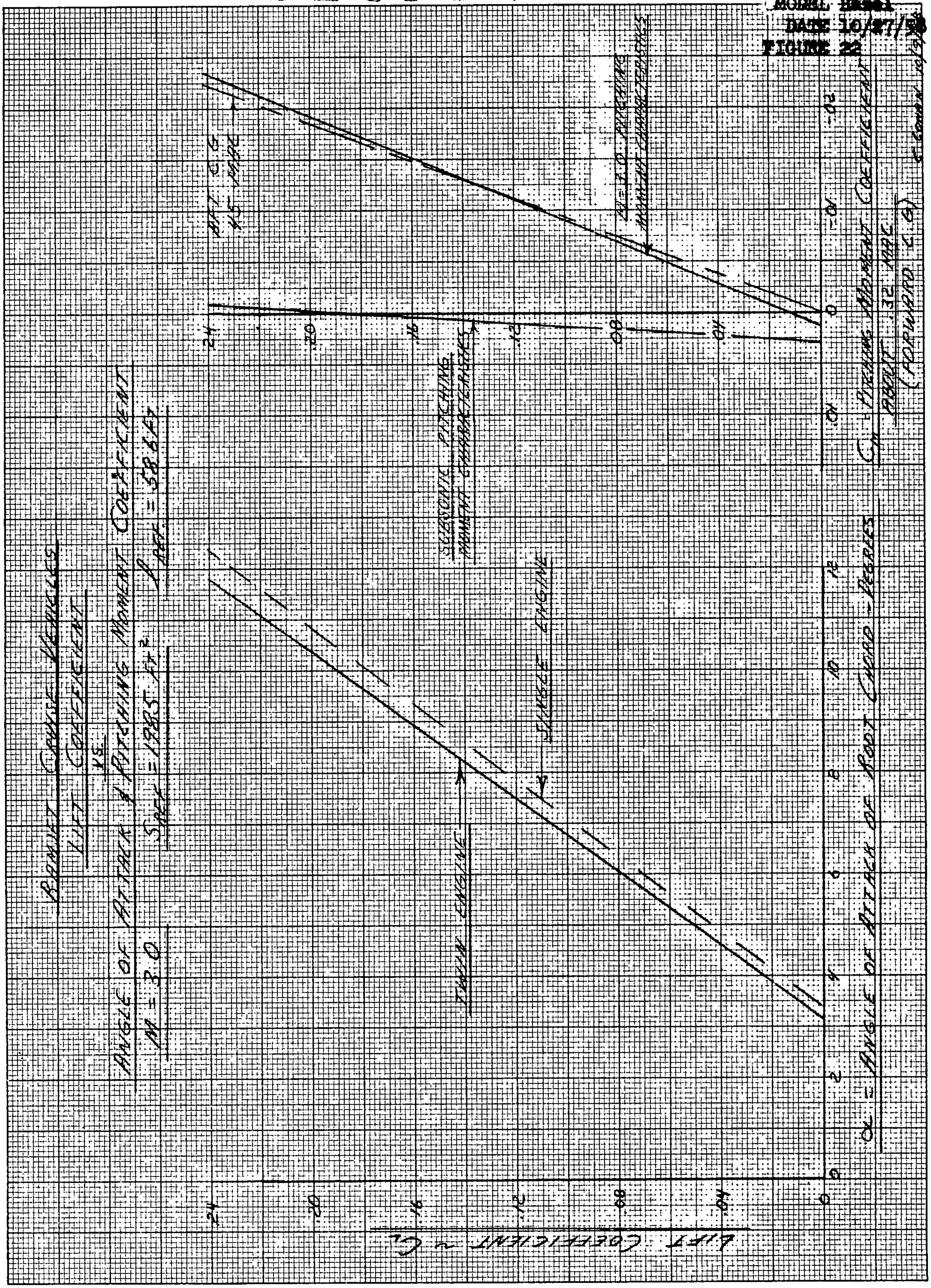
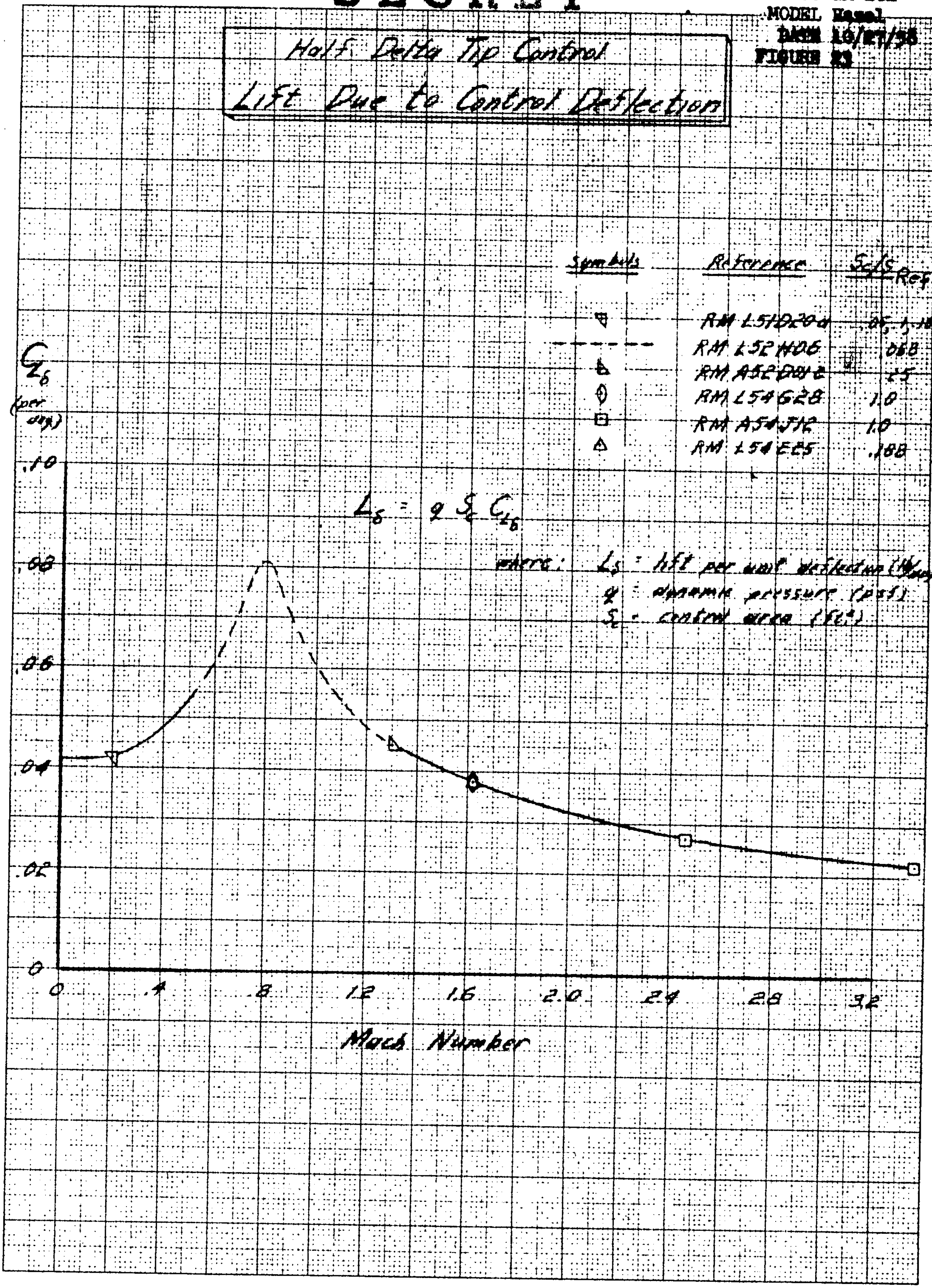


SECRET

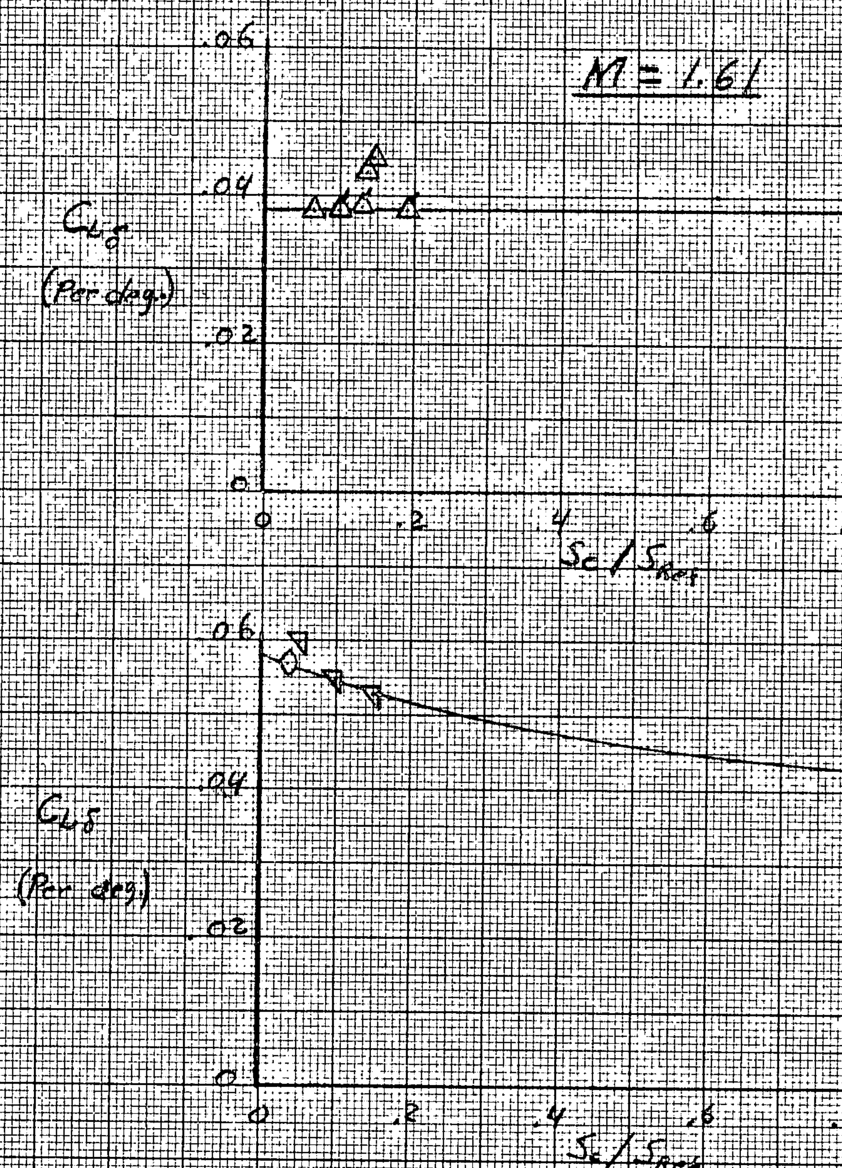

Half Delta Tip Control
Lift Due to Control Deflection
Symbols
Reference
S_c/S_{Ref}
RM L51D20a
RM L52H06
RM A52D02
RM L54G28
RM A54J12
RM L54E25
.068
1.0
1.0
.188
C_{L_δ} (per deg)
$L_\delta = q\,S_c\,C_{L_\delta}$
where: L_δ = lift per unit deflection (lb/deg)
q = dynamic pressure (psf)
S_c = control area (ft²)
.10
.08
.06
.04
.02
0
0
.4
.8
1.2
1.6
2.0
2.4
2.8
3.2
Mach Number

HALF DELTA TIP CONTROLS

C_{L_δ} vs. CONTROL SIZE

Note: C_{L_δ} is based on Control area, S_c

Symbols	Reference	Control
◊	RM L54G28	Complete Wing
△	RM L54E25	tip
△	"	Trailing Edge
◊	CVAL 101G	tip
▽	RM L51D20a	tip



Half Delta Tip Control
Control Effectiveness & Complete Wing a.c.

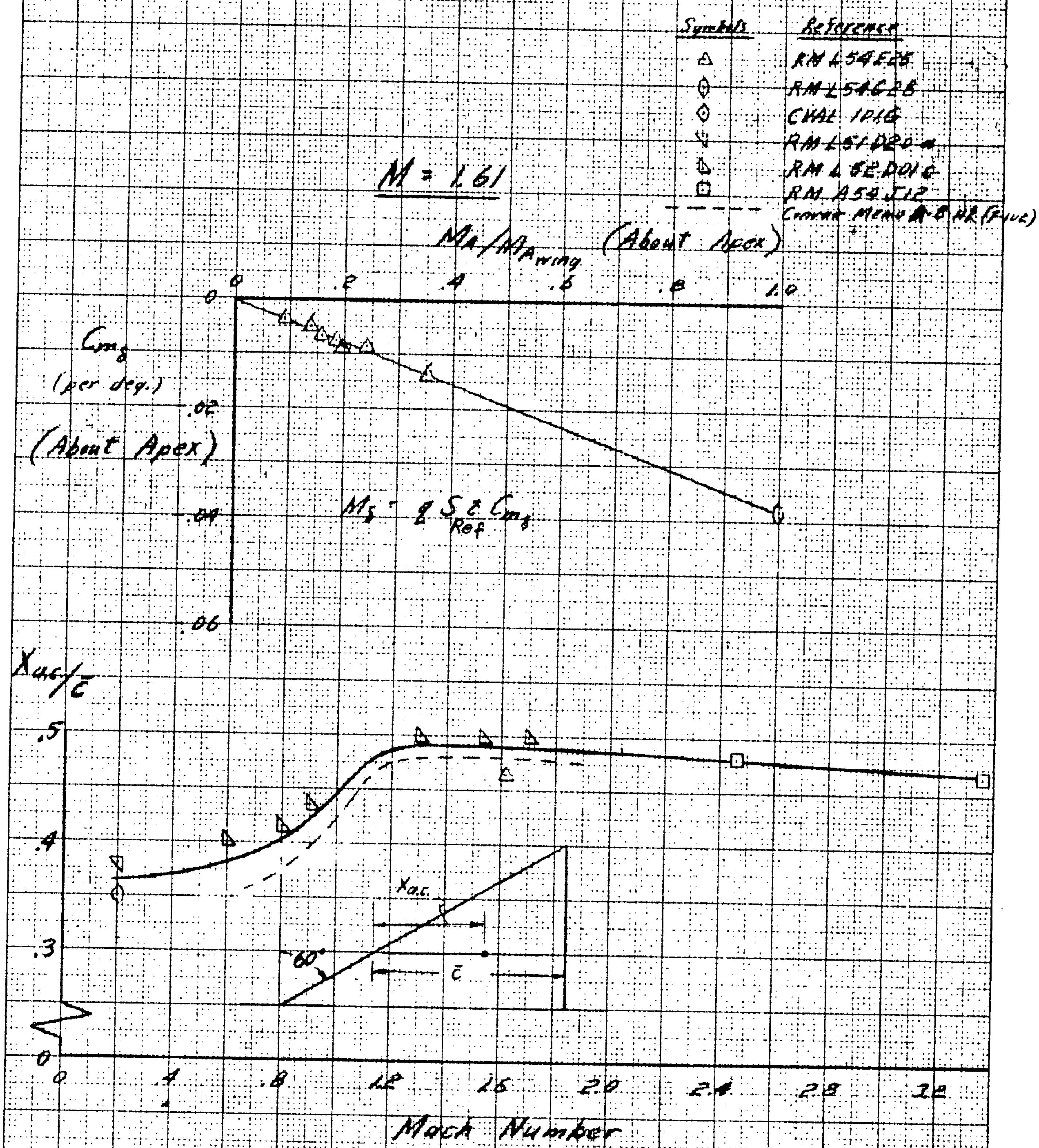


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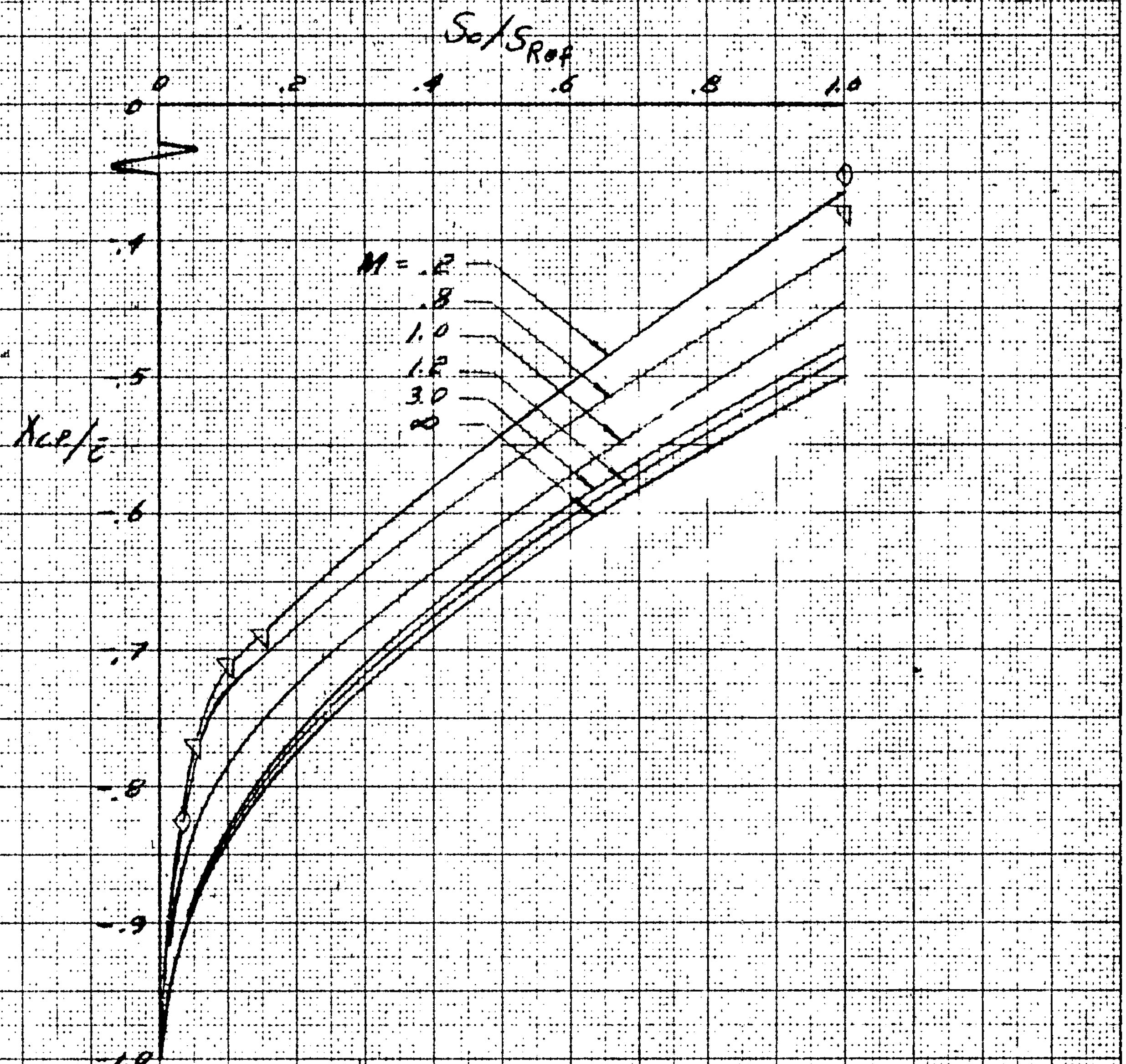
FIGURE 26

Half Delta Tip Control
Control Effectiveness

Note: $X_{c.p.}$ is measured from leading edge of MAC to the center of pressure of the additional load due to control deflection



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MODEL Hazel
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FIGURE 27

Half Delta Tip Control
Control Force & Center of Pressure

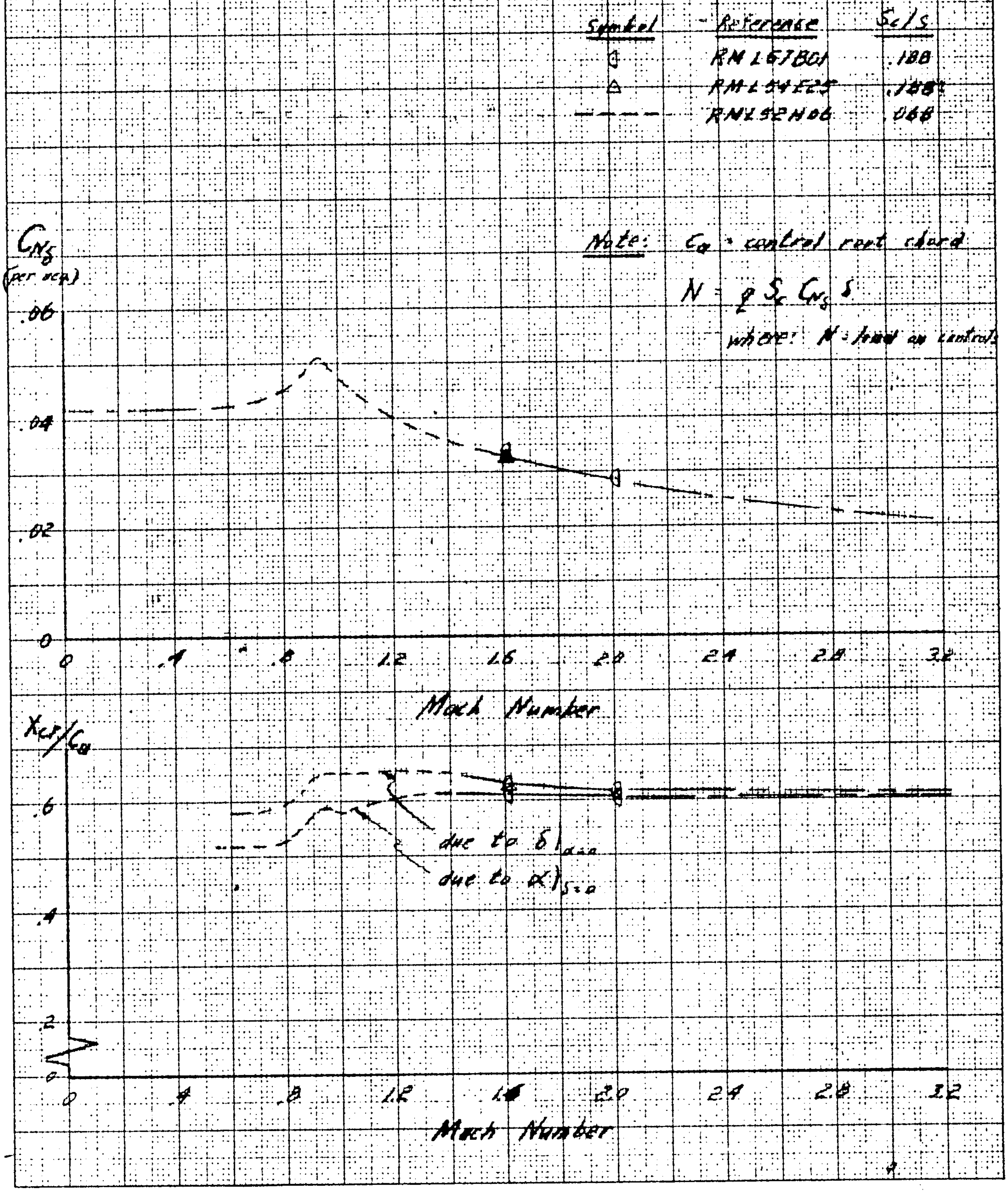

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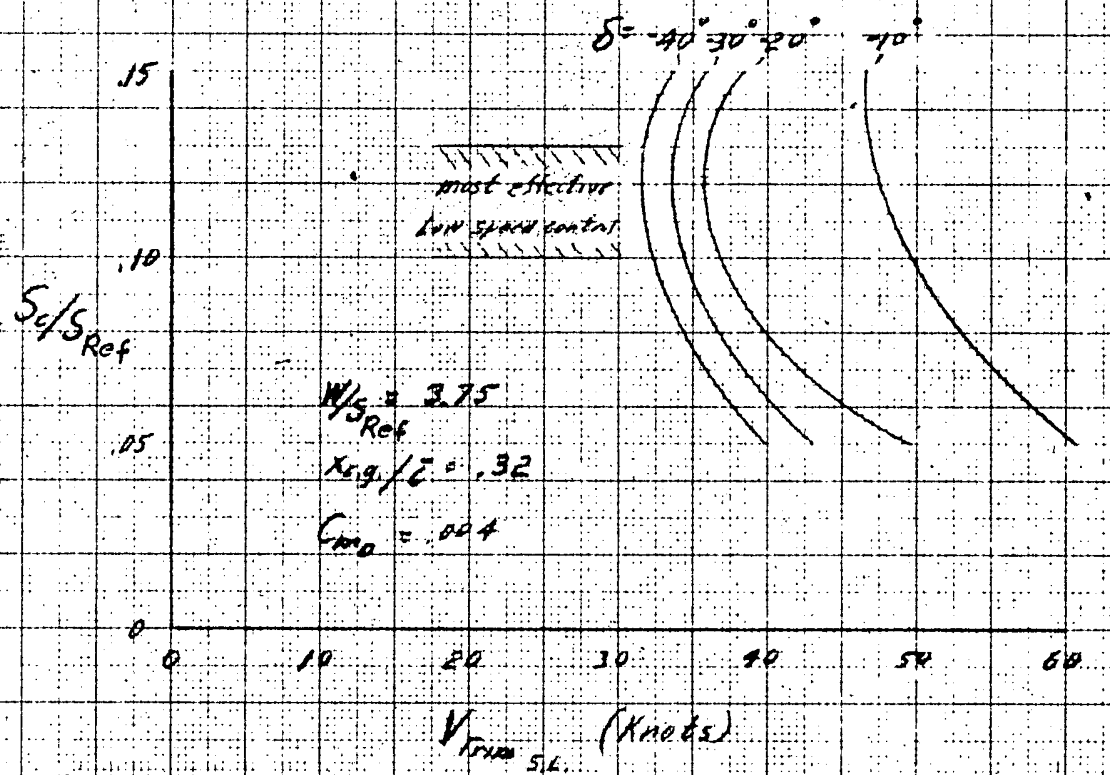
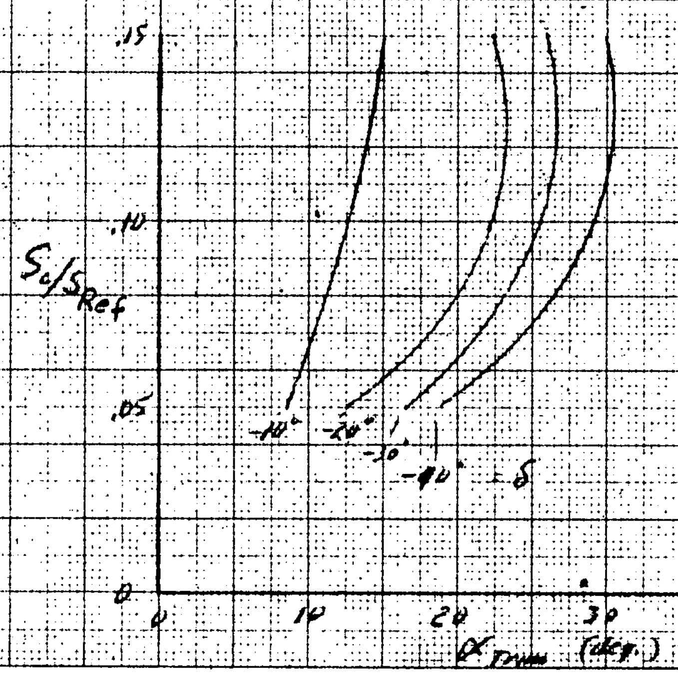
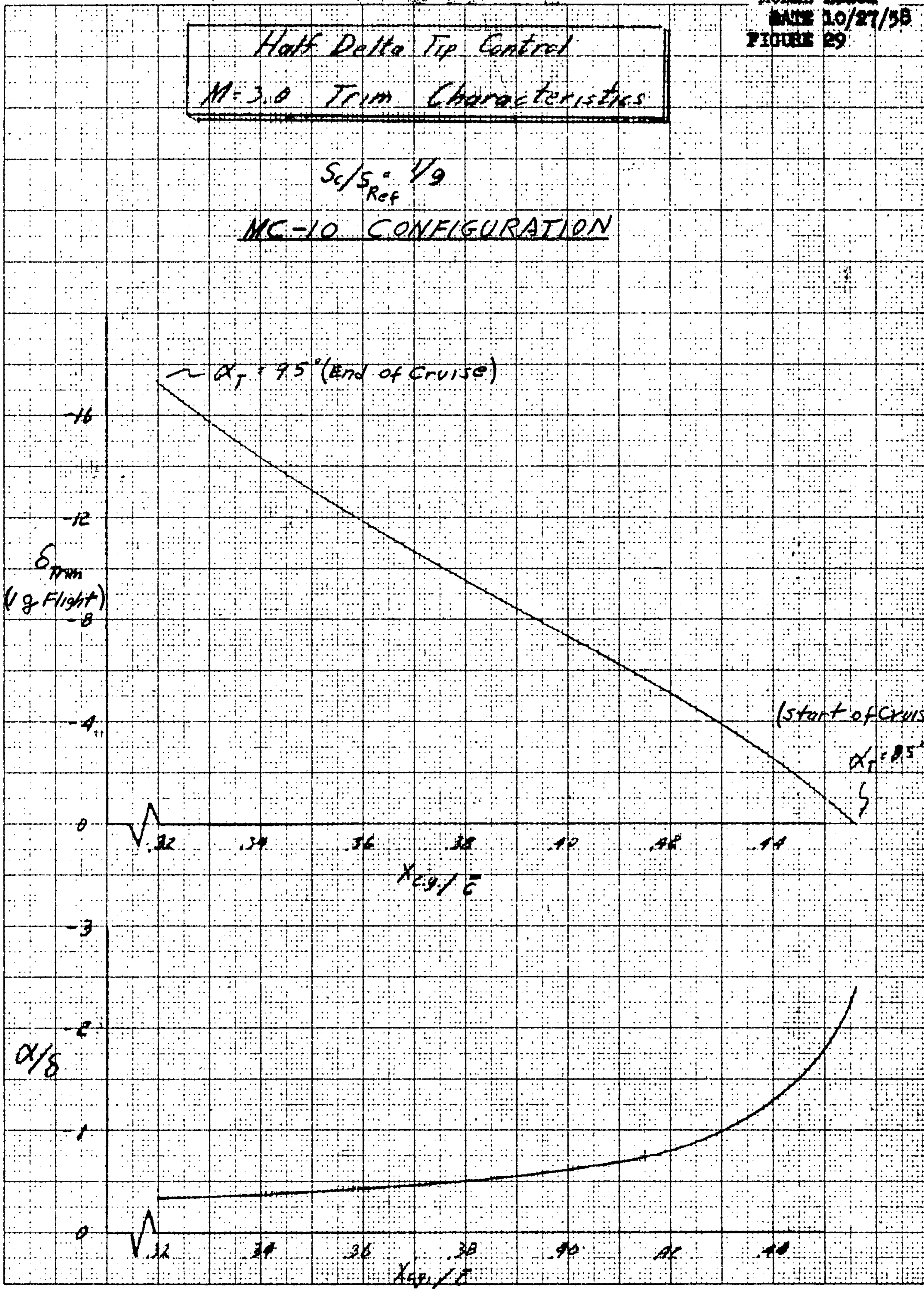
FIGURE 28

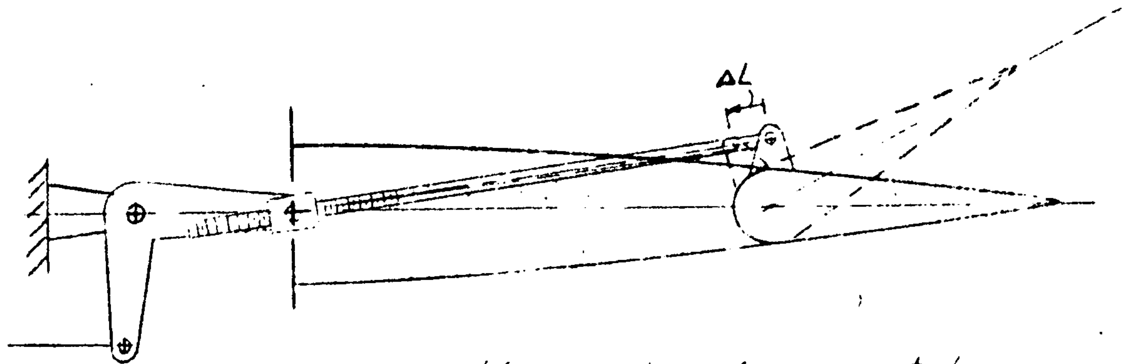
Half Delta Tip Control
Low Speed Trim Characteristics



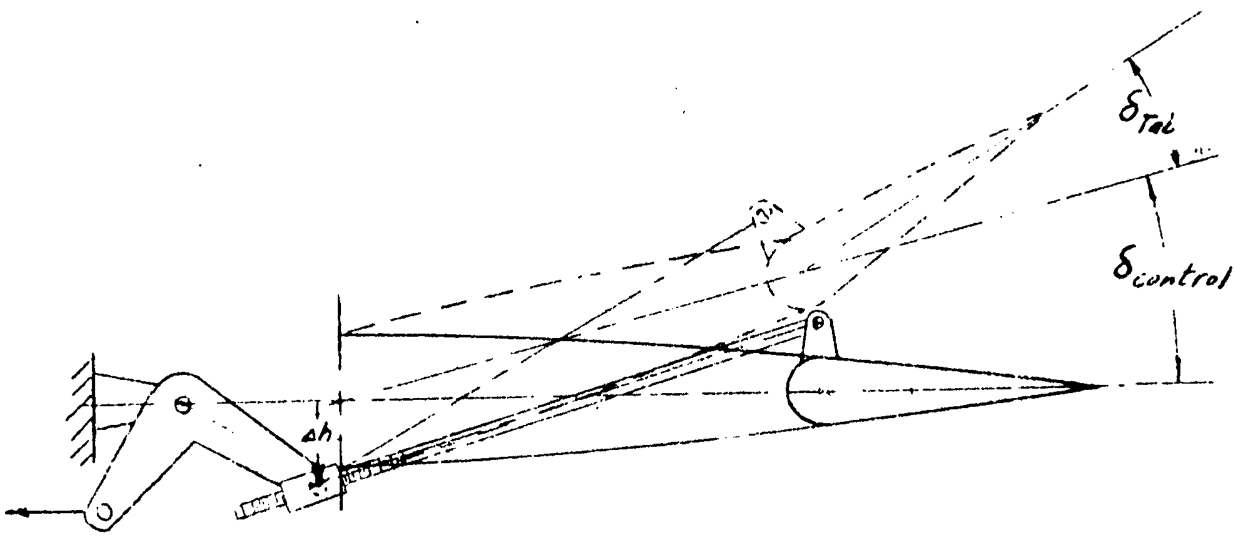



Half Delta Tip Control
M=3.0 Trim Characteristics
$S_c/S_{Ref} = 1/9$
MC-10 CONFIGURATION
$\alpha_T = 9.5°$ (End of Cruise)
(start of Cruise)
$\alpha_T = 8.5°$
-16
-12
δ_{Trim}
(1 g Flight)
-8
-4
0
.32
.34
.36
.38
.40
.42
.44
$X_{c.g.}/\bar{c}$
-3
-2
α/δ
-1
0

Trim + Anti-Balance Tab
Schematic Description



Trim - obtained by changing tab push rod length



Anti-Balance - obtained by changing tab push rod "fixed" end position

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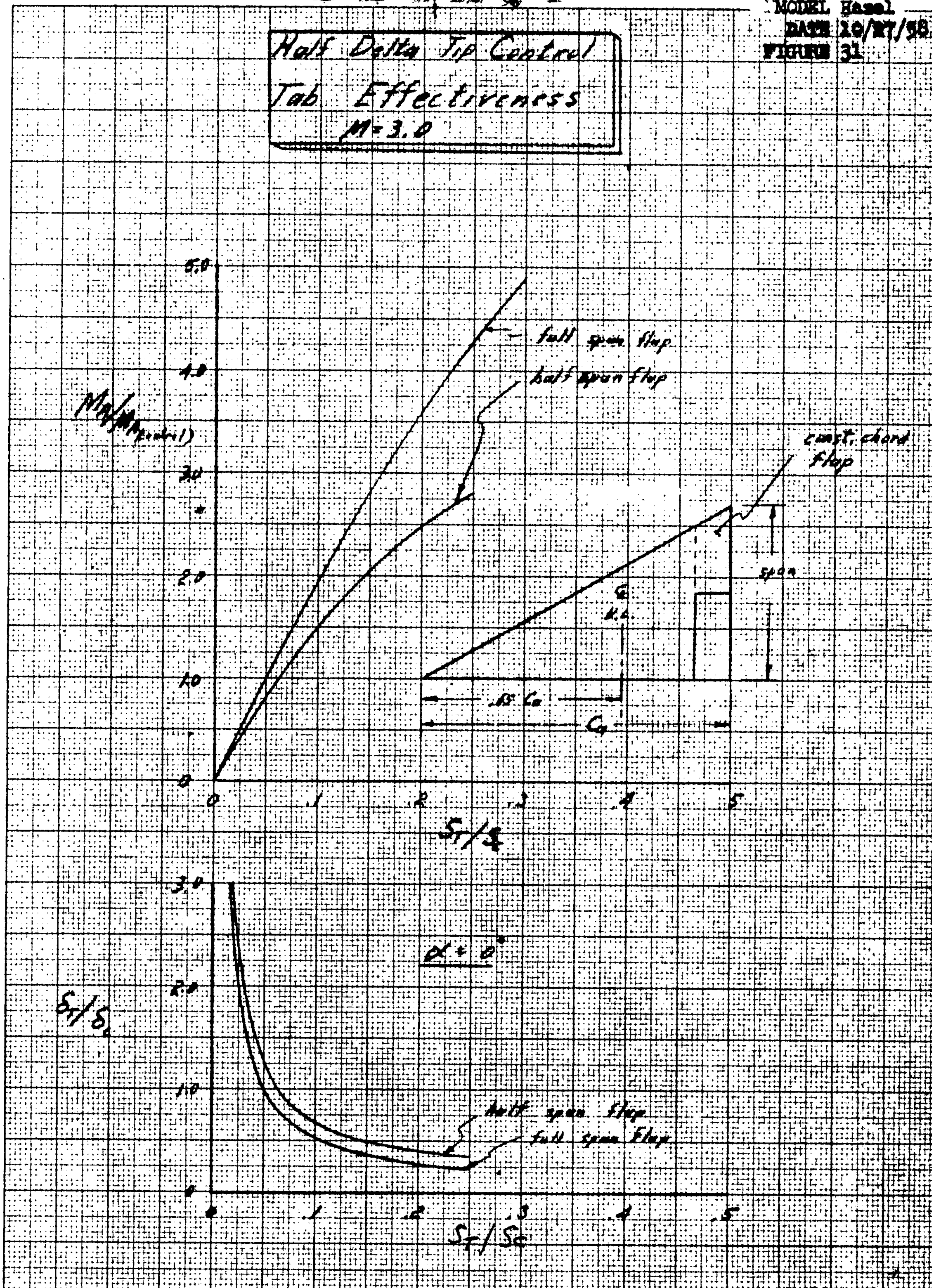
FIGURE 31



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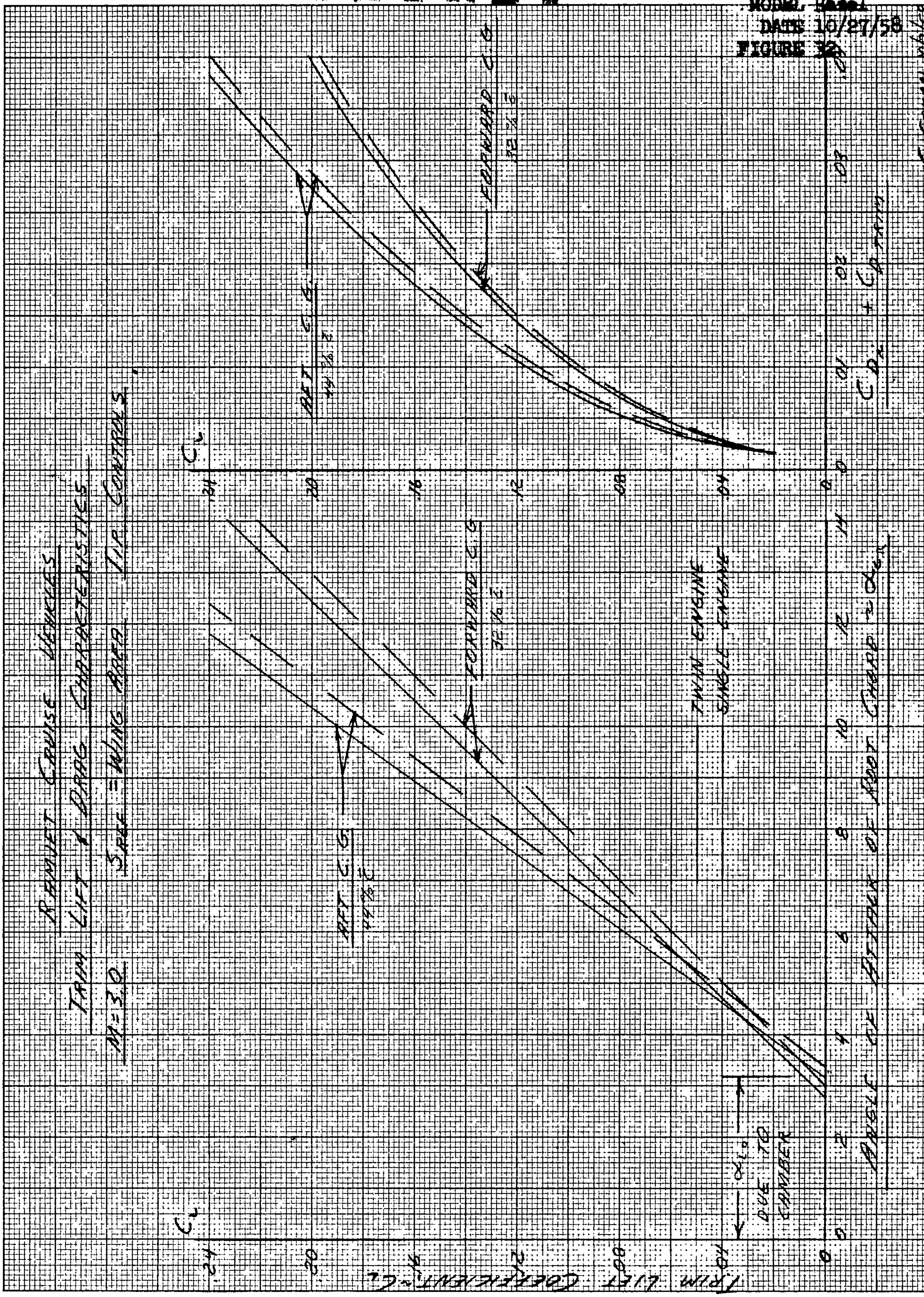

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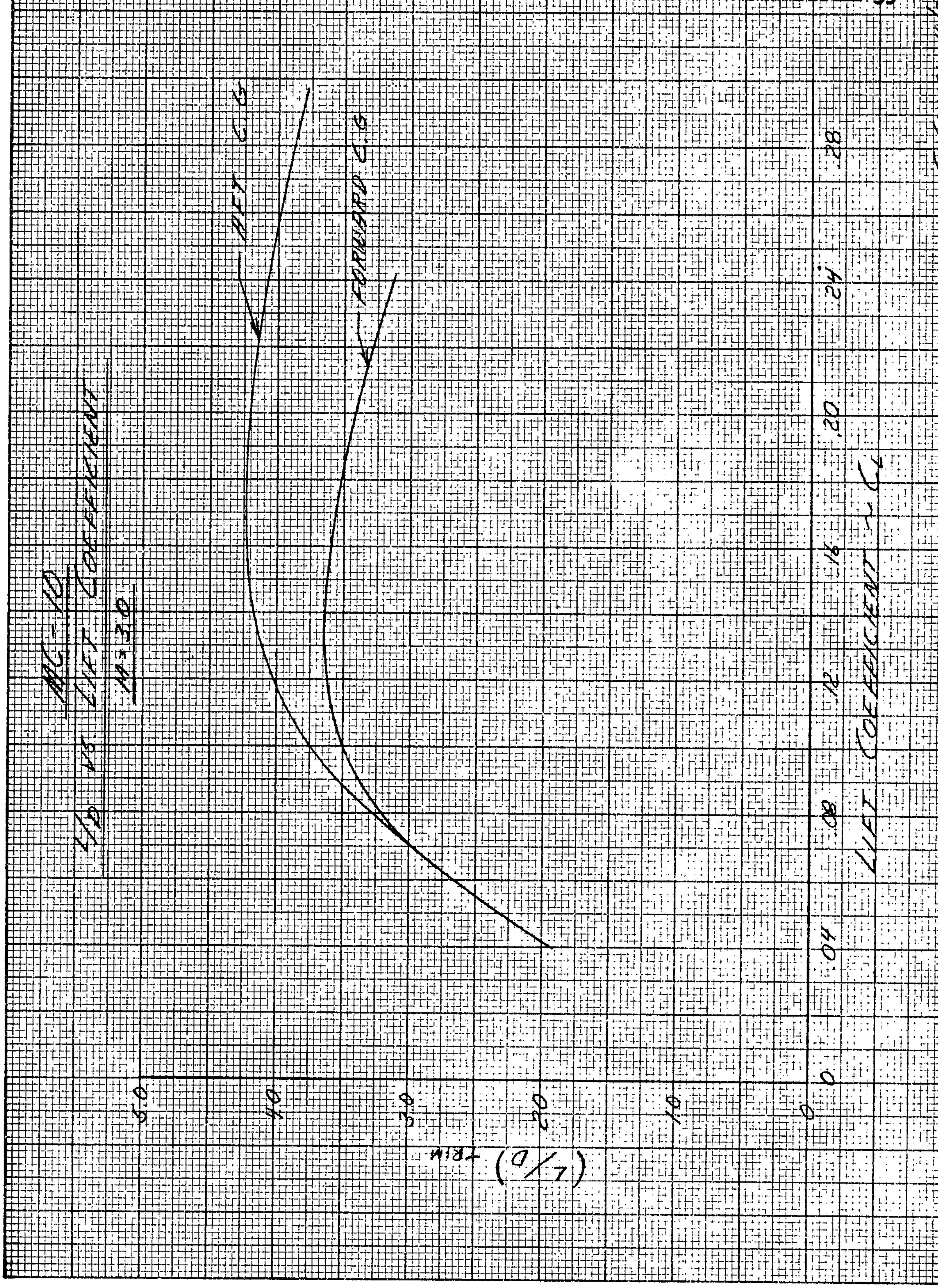
FIGURE 33



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SECRET

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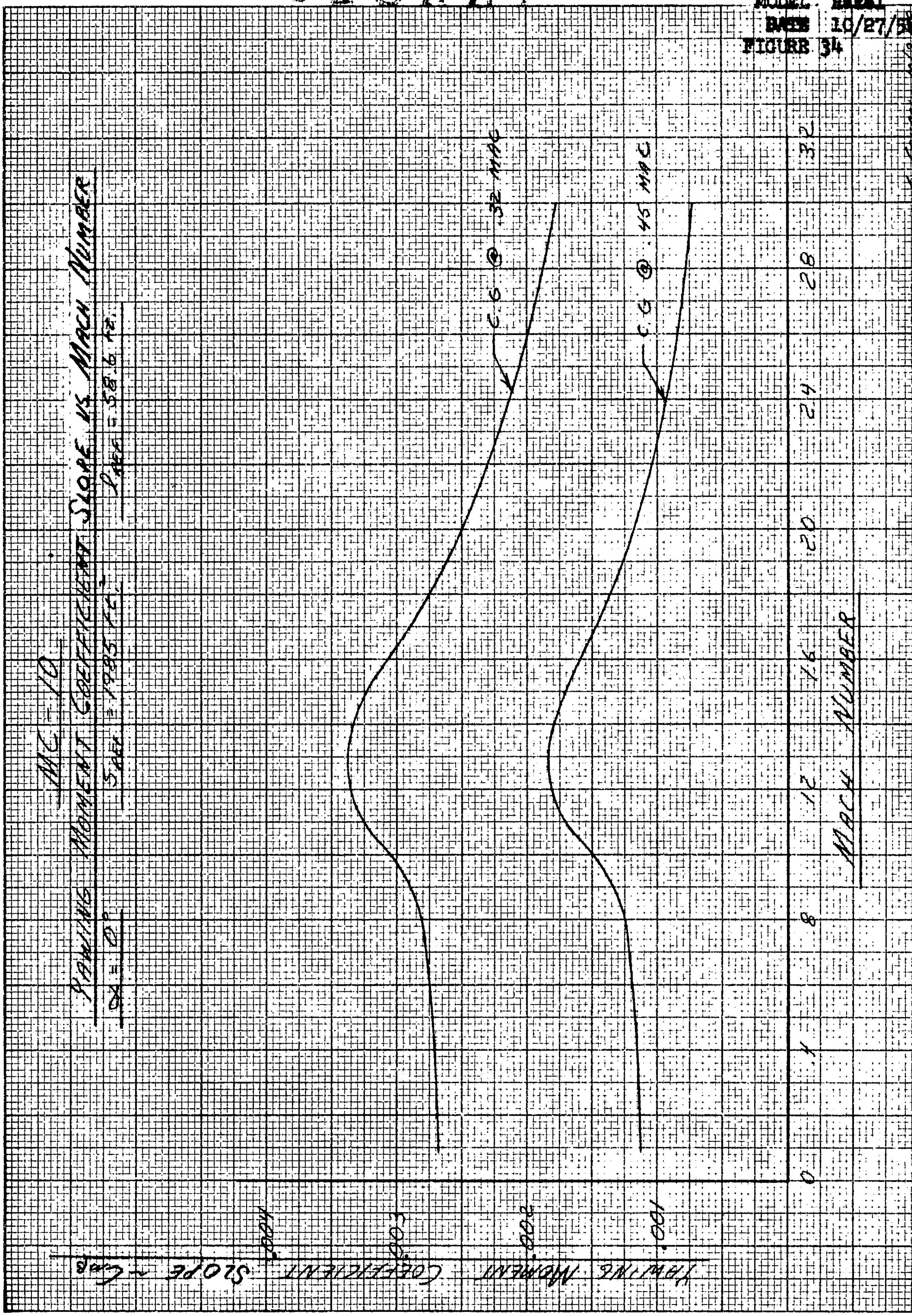
FIGURE 34



SECRET

SECRET

SECRET

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